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AIRPLANE ACTUATION TRADE STUDY

Rockwell International Corporation
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Prepared for
AIR FORCE FLIGHT DYNAMICS LABORATORY
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Wright-Patterson Air Force Base, Ohio 45433

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This report contains the results of the Airplane Actuation Trade Study Program. The study, conducted in three phases, included establishment of actuation requirements; design of two airplanes (a baseline airplane and an all-electric airplane) and a trade study of the two airplanes plus several minor variants. The trade study includes quantitive comparison data relative to weight, reliability, maintainability, and life cycle cost.

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20. ABSTRACT (continued)

The study results indicate that the "All-Electric" approach did not provide a viable alternative to the more conventional "Hydraulic Electric" approach as these approaches would be developed and applied to aircraft of the mid-1990 time period.

FOREWORD

This Final Technical Report summarizes the work accomplished through June 1982 by Rockwell International's North American Aircraft Operations on Air Force contract F33615-79-C-3615, Airplance Actuation Trade Study. The contract was initiated under Project Number 2403 entitled "Flight Control System Development". The USAF Project Engineers for the trade study are Greg Cecere of AFWAL/FIGL and Kenneth Binns of AFWAL/POOS. This report covers work performed between 25 June 1979 to 1 July 1982.

The principal contributors to this activity at Rockwell International were C. W. Helsley Jr., A. Davanzo and W. Frantz.

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INTRODUCTION

1.1 Status - Current aircraft are characterized by two main forms of on-board secondary power generation distribution, and utilization, i.e., electrical power and hydraulic power. In general, hydraulic power is generated, distributed, and utilized for the majority of the high power output actuation functions such as primary and secondary flight control surfaces, landing gear extension and retraction, brakes, nose wheel steering, etc. and electrical power is used for everything else. This division of functional responsibility developed over the years, largely as a result of the ever increasing demands of high performance aircraft for higher levels of controllability in the presence of high "G" forces, thus making it necessary to amplify pilot forces with power actuators. The accepted fact that hydraulic actuators have enjoyed many advantages over electromechanical actuators for high torque, high horse power application shifted the pendulum in their direction. In the interim, since hydraulic actuation was accepted approximately twenty five years ago, many changes have occurred. These changes are discussed more fully in paragraph 1.3 but they lead up to the objectives of this study which are stated here: "Establish advantages/disadvantages and life cycle cost impact of hydraulic actuation and power-by-wire actuation of aircraft in the 1990 + time frame." A secondary objective of this effort was to identify technology needs and development requirements for future aircraft actuation systems.

1.2 Scope - This program was conducted to satisfy the objectives listed above under the following guide lines.

1.2.1 Conduct a trade-off study between a power-by-wire actuation airplane and one that retains an engine-driven hydraulic syste for actuation. NOTE: A power-by-wire actuation airplane was defined as either (1) removal of all engine driven hydraulic pumps and hydraulic power distribution systems and replacement with electrical power generation and distribution systems to the actuator location where electrical power was then converted to hydraulic power for actuation or (2) same as above except that the electrical power was converted to mechanical power for actuation directly or (3) some combination of (1) and (2). Hybrid systems retaining engine driven pumps for specific functions were considered viable options.

1.2.1.1 Use the ATS concept as the point of reference airplane on which the trade study was to be conducted.

1.2.1.2 Use the 1990+ time frame as a design reference for all system options included above.

1.2.1.3 Include other utility functions such as environmental control systems if they became relevant to the basic trade.

1.2.1.4 Assess the trade on the basis of performance, reliability, maintainability, weight, life cycle costs, growth potential, survivability, and environmental constraints.

1.3 Background - The advent of jet engines in the early fifties greatly increased the performance of military aircraft and made it necessary to supplement pilot control forces with power amplification (actuators) at the control surfaces. At the time these actuators became necessary there was, as there is now, two possible power choices, electrical or hydraulic. As the weight and space penalties were examined to make the choice, i.e., hydraulic or electrical actuation, there was no real contest. Hydraulic actuation was clearly superior, if not indispensable. The development of hydraulic actuation had reached the point where primary control surface actuation had become synonymous with hydraulic actuation. Nor was hydraulic actuation limited to primary control surfaces; landing gear retraction and extension,

1.3 (cont.)

brakes, flaps and slats operation,, and auxiliary functions such as nose wheel steering, etc. were all conventionally done with hydraulic actuation on most airplanes. The development of engine driven hydraulic power systems to service these actuation needs had therefore also evolved and matured over the years and had reached a high degree of refinement. However, looking down the road to future design, several significant factors were emerging which suggested that primary flight control actuation, as well as other actuation functions, need not continue to be done in the conventional manner.

Perhaps the single greatest factor that was stimulating the need to examine power-by-wire actuation, as an alternative to conventional hydraulic actuation, was the increasing importance of avionics and in particular fly-by-wire. Fly-by-wire signal transmission dictates absolute electrical power reliability as a paramount design requirement for future airborne electrical power systems. There was, and is, a direct conflict between the independent redundancy required for electrical power support of fly-by-wire and the longstanding independent hydraulic power redundancy requirements to support flight control actuation. For example, on a single engine airplane, with a four channel fly-by-wire system, if carried to the extreme, this approach could have resulted in four engine driven electrical generators plus the normal two hydraulic pumps.

A second factor was the mounting cost to design, develop and maintain the engine driven hydraulic power and distribution systems that were being implemented to utilize the generally admirable qualities of hydraulic actuators. These hydraulic systems were plaged with leaks, contamination, flammability and generally high life cycle costs, particularly maintenance costs.

A third factor concerned the credibility of power-by-wire actuation as an alternate to conventional hydraulic actuation, i.e., hydraulic actuators supplied by engine driven hydraulic systems. In the twenty-five years since the adoption of hydraulic actuators, several technologies in the electrical and electromechanical area had emerged which warranted a reinvestigation of electromechanical actuation for application to primary flight control and other actuation systems. Some of these advancements were high voltage power supplies, permanent magnet motors using rare earth magnets, electronic commutation and an improvement in solid state power switching devices.

2.0 STUDY AIRCRAFT DEFINITION

2.1 Baseline Air-to-Ground Tactical Fighter Requirements - An integrated baseline set of 1990 tactical air-to-ground fighter mission requirements was selected and is presented in this section. This set of baseline requirements provided a foundation and framework within which the trade studies could be conducted. The selection was based on recent studies (reference 31 through 33) and represent a very likely set of requirements for the time period.

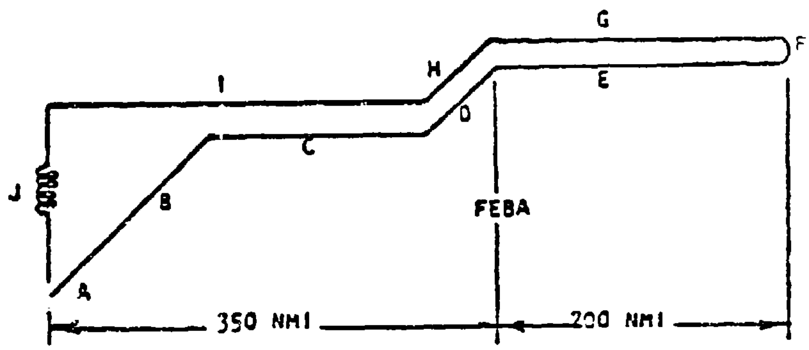
2.1.1 Design Mission Profile Requirements - The baseline "most probable" 1990's advanced tactical mission profile requirements turned out to be a high altitude supersonic design mission with a mach 2.0 class penetration speed supplemented by an alternate low level terrain following profile capability in the high subsonic speed category. The specifics of the design sizing profile and the alternate capability profile are presented in figures 2-1 and 2-2. The mach 2.2 penetration for the high altitude profile was selected because it represented the approximate upper speed boundary for use of the composite materials generally expected to be employed in the time period. The mach 0.9 penetration selected for the alternate capability low level mission was selected as a reasonable compromise between higher speeds providing better survivability and lower speeds providing better target acquisition. The two penetration design points selected also presented moderately high requirements to the aircraft actuation system design in terms of hinge moments, temperature environment and actuation rate requirements. Thus, these baseline design profiles, while representing expected future mission requirements, also presented moderate challenges to actuation system technology without preselecting a particular type of actuation system through selection of extreme combinations of requirements. The remaining requirement particulars indicated on figures 2-1 and 2-2 (distances, payload, combat allowances, takeoff and landing distances, etc.) were selected as nominal representative values from recent industry/government studies (reference 31 through 33). These latter factors influenced size of the aircraft but had only secondary influence on secondary power and actuation system concepts, design and technology requirements.

Other mission profiles considered prior to selection of the mach 2.0 class aircraft were; mach 3.0 high altitude penetrator, 0.7 low level penetrator, and mach 1.2 low level penetrator.

2.1.2 Additional Design Criteria - The following paragraphs present additional design criteria that have significant effects on actuation system designs. They have been derived from the same industry/government studies (reference 31 through 33) that established the basic mission profiles and were selected for their compatibility with the specific basepoint design mission profiles.

2.1.3 Service Life and Usage Criteria - The total service life requirement for this type of aircraft was 8000 total flight hours, based on current DOD policy to maintain a major aircraft in the inventory for 15 years plus the assumption of one major short war. The combat usage consisted of 60 flights of each of the combat profiles illustrated in figures 1 and 2.

Training usage was developed around the philosophy of providing adequate flight time to complement a significant ground simulator training program. Limited actual supersonic flight and low-level terrain following were scheduled to



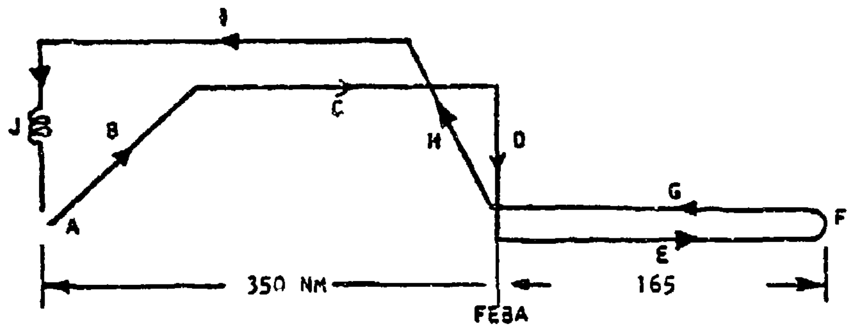
T.O. & LDG DISTANCE: ≤3000 FT GND ROLL   MISSION RADIUS - 550 N MI

(A/G + A/A WEAPONS)   WEAPONS PAYLOAD - 5,680 LBS
(A/G WPNS + FAIRINGS)   PAYLOAD DROPPED - 5,030 LBS

MISSION SEGMENT	DISTANCE (N MI)	MACH	ALTITUDE (FT) end	TIME (MIN)
A. TAKEOFF AND ACCEL TO INITIAL CLIMB VELOCITY	0	0-0.81	0	15.8
B. CLIMB AND ACCEL TO CRUISE CONDITION	38.0	0.81-0.9	31,500	4.2
C. CRUISE-OUT	244.2	0.9	32,800	27.8
D. CLIMB AND ACCELERATE TO DASH CONDITION	67.8	0.9-2.2	59,000	3.9
E. DASH TO TARGET	200	2.2	60,100	9.5
F. COMBAT ALLOWANCE*	0	2.2	50,000	2.8
G. DASH - TARGET TO INITIATION OF RETURN DESCENT/DECEL	200	2.2	62,300	9.5
H. RETURN DESCENT/DECEL	185.7	2.2-0.9	39,600	13.3
I. RETURN CRUISE	164.3	0.9	40,300	19.0
J. LANDING/LOITER	0	0.4-0	0	20.0

* COMBAT ALLOWANCE
360° TURN AT $P_S = 0$, MAX A/B
2.2M/50,000 FT

Figure 1. Basepoint High Altitude Design Profile Performance



T.O. & LDG DISTANCE: ≤3000 FT GND ROLL MISSION RADIUS - MAX ACHIEVABLE

(A/G + A/A WEAPONS) WEAPONS PAYLOAD - 5,680 LBS
(A/G WPNS + FAIRINGS) PAYLOAD DROPPED - 5,030 LBS

MISSION SEGMENT	DISTANCE (N MI)	MACH	ALTITUDE (FT)	TIME (MIN)
A. TAKEOFF AND ACCEL TO INITIAL CLIMB VELOCITY	0	0-0.81	0	0.8
B. CLIMB AND ACCEL TO CRUISE CONDITION	38	0.81-0.90	0-31500	4.2
C. CRUISE-OUT	312	0.90-0.90	31500-33000	35.50
D. DESCENT TO DASH CONDITION	0	-	33000-0	0
E. DASH TO TARGET	165	0.9	0.0	16.6
F. COMBAT ALLOWANCE *	0	0.9	0.0	0.45
G. DASH - TARGET TO INITIATION OF RETURN CLIMB	165	0.9	0.0	16.6
H. RETURN CLIMB	26	0.9-0.9	0.0-39000	2.8
I. RETURN CRUISE	324	0.9-0.9	39000-40200	37.5
J. LANDING/LOITER	0	0.38	0	20.0

* COMBAT ALLOWANCE:

360° TURN AT $P_S = 0$, MAX A/B
0.9M/0.0 FT

BCM - BEST CRUISE MACH
BCA - BEST CRUISE ALTITUDE
BLM - BEST LOITER MACH

Figure 2. Basepoint Alternative Performance Evaluation Mission

minimize structural temperature and fatigue design requirements, however, adequate airborne experience was accumulated in high- and low-level missions, navigation, inflight refueling weapon delivery tactics, etc. Allowances were included for the extra landings due to touch-and-go practice and routine around-the-field maintenance checkout flights, etc. Six basic training mission profiles were developed based on current tactical fighter training schedules, modified as appropriate for the capabilities and tactical employment envisioned for the advanced fighter. The field-go-around profile was added to the other profiles or conducted independently to represent touch-and-go training activities or short maintenance checkout flights, etc. The basic mission characteristics are illustrated in figure 3.

The average number of flights flown per aircraft each year on each of these profiles were:

Mission Profile	Avg. No. of Flights/Yr.
Ground attack tactics	54.2
Mission support	21.0
Low-level strike with refueling	20.1
High-level strike	16.9
High-level strike with refueling	12.3
Supersonic combat profile	2.4
Extra field-go-around plus landing	153.1
	280.0

The flight operations indicated above impose 4320 ground-air-ground cycles on the average aircraft for the specified usage. The flight-hour usage accumulated in each of the mission legs illustrated by the totality of flight operations indicated in figure 3 plus the combat operations was the basis for projecting the structural fatigue life spectra, thermal environment design criteria and secondary power and actuation system component duty cycle requirements. Estimated vehicle total lifetime hours usage is shown on table 1.

Based on the usage presented in table 1, a single composite mission was developed which if repetitively flown would produce approximately the same cumulative individual leg usage as noted in the table. This composite mission profile is presented in figure 4. Use of a single design mission of this type facilitated development of detailed design criteria for the aircraft usage. Because aircraft actuation system component duty cycles are influenced by the maneuvering and gust upset restoring requirements by mission leg, the load factor spectra was developed based upon the data of MIL-A8866B for the cumulative life of the aircraft as flown over the composite mission. Table 2 presents this design data. The spectrum includes the effects of long-term, peacetime training usage and an allowance for a representive high-intensity, short-duration wartime employment. The combined flight and ground-air-ground cycle design spectra resulting from this usage is presented as a flight-by-flight composite mission load spectrum containing an appropriate amount of flight time in each of the mission legs in table 1. The composite mission has a duration of slightly over 2.7 hours. A total of 2,935 such missions with an extra field-go-around and landing on every other flight provides the total design life usage.

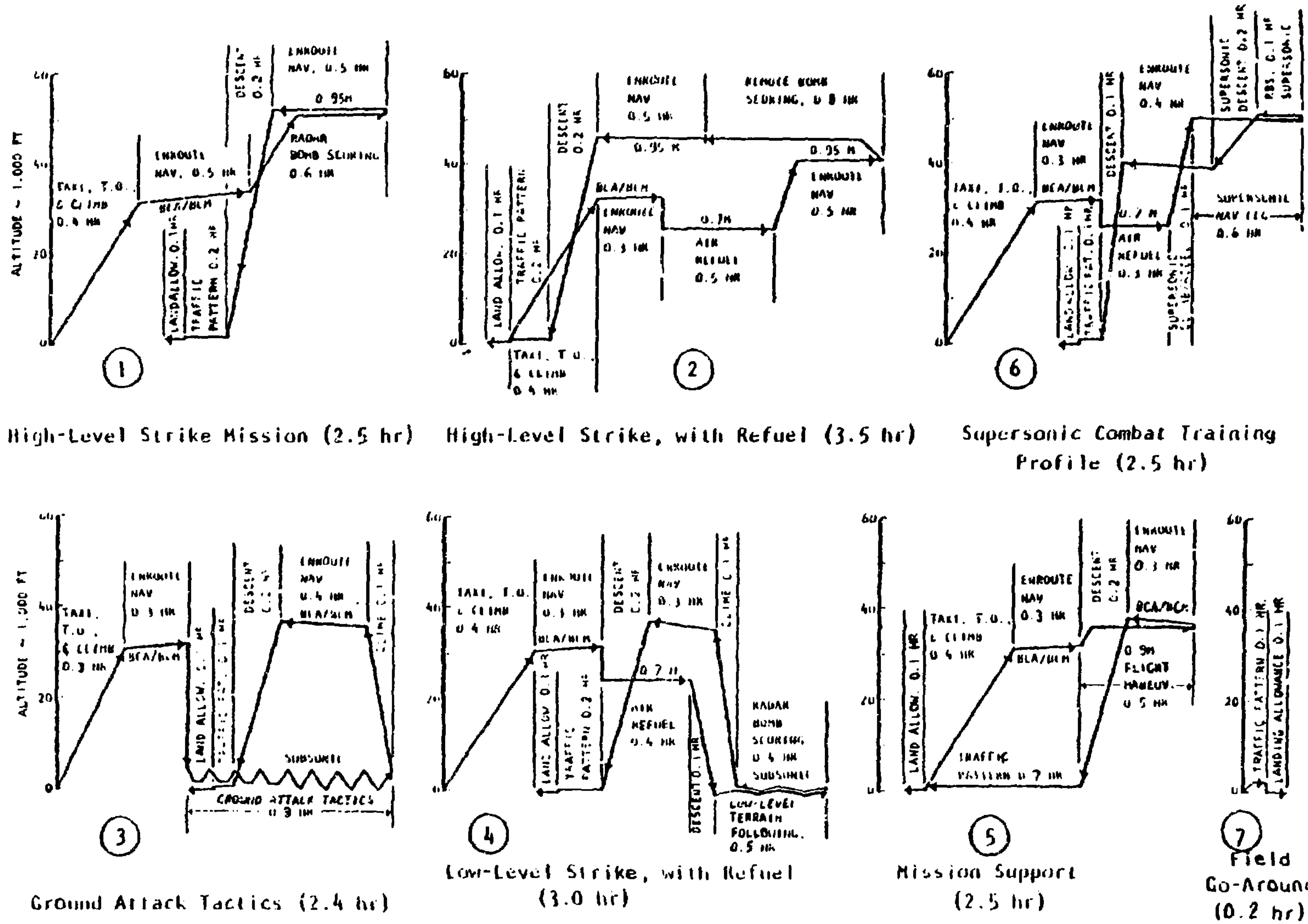


Figure 3. Peacetime Training and Support Missions

TABLE 1. VEHICLE TOTAL LIFETIME HOURS USAGE

MISSION SEGMENTS	SUPERSONIC COMBAT	SUBSONIC COMBAT	PEACETIME MISSIONS* 1 - 5	SUPERSONIC TRAINING	FIELD GO-AROUND	TOTAL HOURS
TAXI, T.O. AND CLIMB	18	30	1150	22		1220
ENROUTE NAV CRUISE	48	60	2561	38		2707
AERIAL REFUEL			315	16		331
LOW LEVEL TERRAIN FOLLOWING		36	402			438
SUPERSONIC ACCEL/CLIMB	(6)			(6)		(12)
GROUND ATTACK TACTICS			1082			1082
SUPERSONIC CRUISE	(24)			(27)		(51)
WEAPON DELIVERY TURN		12				12
FLIGHT MANEUVER (LOITER)		6	234			240
SUPERSONIC WEAPON DELIVERY	(12)					(12)
SUPERSONIC DESCENT	(6)			(11)		(17)
SUBSONIC DESCENT	6	12	596	5		619
TRAFFIC PATTERN (LOITER)	6	6	665	6		683
LAND AND TAXI	6	6	277	5		294
T.O. AND CLIMB					24	24
TRAFFIC PATTERN (LOITER)					223	223
LAND AND TAXI					35	35
	132 (48)	168	7282	136 (44)	282	8000 (92)

* See figure 3

NOTE: Supersonic hours enclosed in parentheses

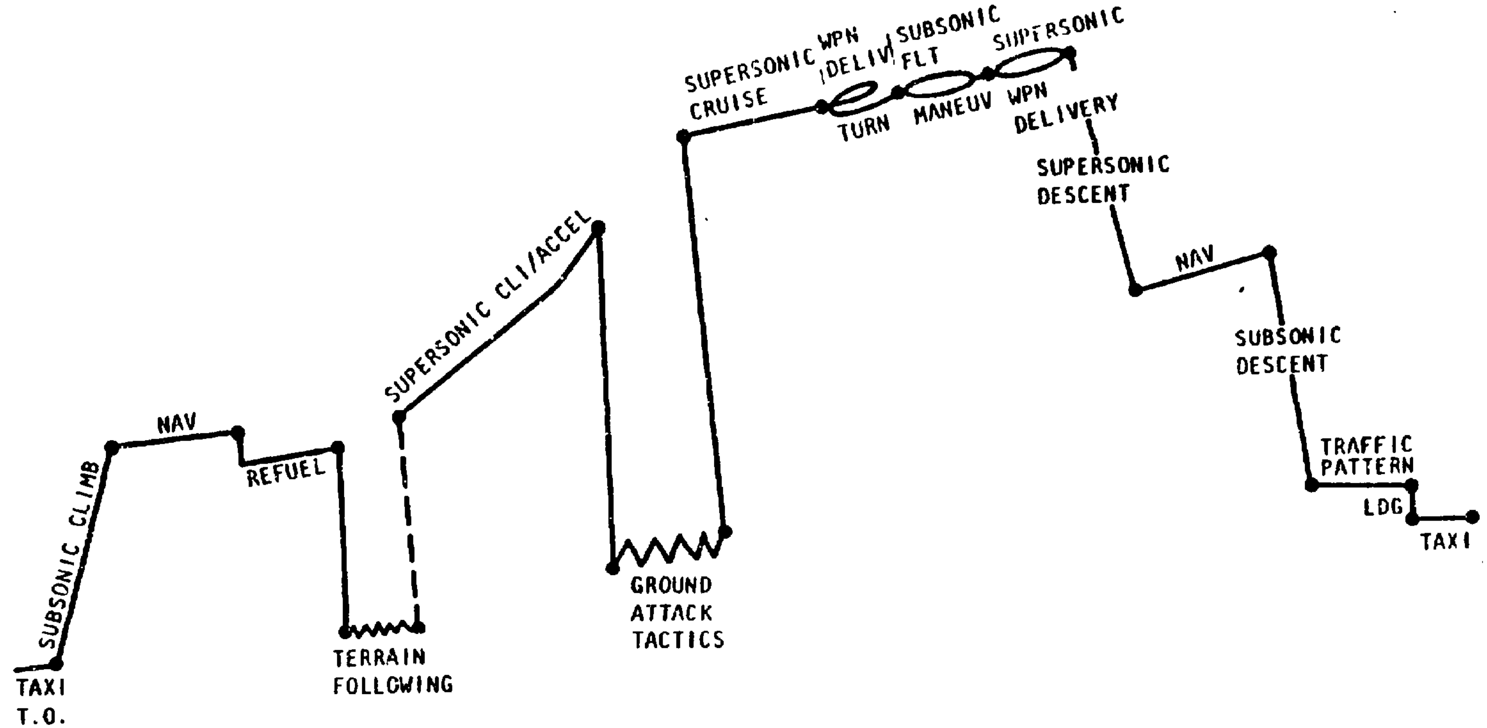


Figure 4. Composite Lifetime Usage Mission

TABLE 2. BASEPOINT FLIGHT-BY-FLIGHT COMPOSITE LOAD FACTOR SPECTRUM (COMBINED COMBAT AND TRAINING MISSIONS)

MISSION SEGMENT	LOAD FACTOR MAX	LOAD FACTOR MIN	LOAD FACTOR CYCLES/MISSION
1. Taxi	1.3	0.7	1
	1.2	0.8	15
2. Takeoff & Climb	3.3	1.0	0.01
	2.7	1.0	0.1
	2.1	1.0	1.1
	1.9	1.0	2
	1.7	1.0	4
	1.5	1.0	8
	1.3	1.0	16
3. Enroute Navigation, Outbound	4.6	1.0	0.01
	4.5	1.0	0.01
	4.2	1.0	0.1
	4.1	1.0	0.1
	3.0	1.0	1
	2.8	1.0	1
	2.5	1.0	2
	2.3	1.0	2
	2.0	1.0	4
	1.7	1.0	4
	1.4	1.0	8
	1.1	1.0	8
4. Aerial Refuel	1.7	0.4	0.01
	1.5	0.6	0.1
	1.3	0.7	1
	1.3	0.7	2
	1.2	0.8	28
5. Low Level Terrain Following	2.2	0.4	0.01
	2.1	0.4	0.21
	2.0	0.4	1
	2.0	0.5	1
	1.9	0.5	4
	1.8	0.5	4
	1.7	0.5	4
	1.6	0.6	8
	1.5	0.6	8
	1.4	0.6	16
	1.2	0.8	16
6. Supersonic Climb/Accelerate	2.3	1.0	0.01
	1.6	1.0	0.1

TABLE 2. BASEPOINT FLIGHT-BY-FLIGHT COMPOSITE LOAD FACTOR SPECTRUM (COMBINED COMBAT AND TRAINING MISSIONS) (CONTINUED)

MISSION SEGMENT	LOAD FACTOR MAX	MIN	CYCLES/MISSION
7. Ground Attack Tactics	7.4	-0.3	0.01
	6.3	0.1	0.1
	5.2	0.4	1
	4.8	0.5	2
	4.3	0.6	4
	3.6	0.7	8
	2.7	0.8	16
	1.7	0.9	27
8. Supersonic Cruise	2.9	1.0	0.01
	2.8	1.0	0.01
	2.6	1.0	0.01
9. Weapon Delivery Turn	7.1	-0.1	0.01
	5.9	0.2	0.1
	4.1	0.5	1
	4.1	0.6	2
	3.4	0.7	4
	2.5	0.8	8
	1.7	0.7	11
10. Flight Maneuver	4.2	1.0	0.01
	4.1	1.0	0.01
	3.1	1.0	0.2
	2.0	1.0	1
	1.9	1.0	1
	1.5	1.0	2
	1.4	1.0	2
11. Supersonic Weapon Delivery	4.4	1.0	0.01
	4.1	0.4	0.01
	3.8	1.0	0.1
	2.3	1.0	1.0
	2.1	0.7	1.0
12. Supersonic Descent	2.3	1.0	0.01
13. Enroute Navigation, Return	4.5	1.0	0.01
	4.0	1.0	0.1
	2.7	1.0	1
	2.1	1.0	2
	1.6	1.0	4
14. Subsonic Descent	4.5	1.0	0.01
	3.9	1.0	0.1
	2.9	1.0	1
	2.4	1.0	2
	1.7	1.0	4

TABLE 2. BASEPOINT FLIGHT-BY-FLIGHT COMPOSITE LOAD FACTOR SPECTRUM (COMBINED COMBAT AND TRAINING MISSIONS) (CONCLUDED)

MISSION SEGMENT	LOAD FACTOR		
	MAX	MIN	CYCLES/MISSION
15. Traffic Pattern	4.8	1.0	0.01
	3.7	1.0	0.1
	2.7	1.0	1
	2.3	1.0	2
	1.9	1.0	4
	1.4	1.0	8
16. Landing	4.3	1.0	0.01
	3.7	1.0	0.1
	2.4	1.0	1
	1.8	1.0	2
17. Taxi	1.3	0.7	1
	1.2	0.8	15

2.1.4 Structural Maneuvering Design Criteria - The structural design load factor requirements were typical standard values for Air Force air-to-ground tactical fighters.

Subsonic	+7.33g	-3.0g
Supersonic	+6.50g	-3.0g

Maximum maneuver roll rate for structural design was 270 degrees per second.

2.1.5 Temperature Design Data - Flight design temperature data were developed to complement the composite design mission leg described above. Table 3 presents the design temperature data. The design standard-day temperatures for each leg of the composite design mission are presented for critical and typical locations on the wing/fuselage structure; e.g., upper and lower surface, 1 foot and 3 feet back from the leading edge. Total stagnation temperature would exist inside the engine inlet duct.

2.1.6 Vibration/Acoustics - Approximate prediction of the vibration/acoustic environment was made using the basepoint aircraft flight envelope and operating characteristics of the propulsion system. Predictions for boundary layer and maximum power engine noise, and weapons bay acoustic environment are shown in figure 5.

2.1.7 Reliability - Air vehicle subsystem reliability, maintainability, and survivability characteristics were considered very important to operating costs and effectiveness of a tactical air-to-ground fighter. Design goals in these areas could significantly affect selection of subsystems concepts arrangements. Overall mission reliability allocation is shown in table 4. The basis for these allocations is reference 4.

TABLE 3. BASEPOINT COMPOSITE MISSION FLIGHT DESIGN TEMPERATURE DATA
STANDARD DAY CRITICAL SOLAR EFFECTS INCLUDED

MISSION SEGMENT	REPRESENTATIVE MACH/ALT	DESIGN TEMP °F TOTAL TEMP	FUSELAGE	WING
1. Taxi	0 / SL	59	59	59
2. Takeoff & Climb				
Low	0.39M/SL	75	74	74
High	0.9M/32K'	12	16	15
3. Enroute Navigation				
Low	0.9M/31K'	16	20	19
High	0.9M/36K'	-6	0	-1
4. Aerial Refuel	0.7M/25K'	12	18	17
5. Low Level Terrain Following	0.9M/SL	143	139	138
6. Supersonic Climb/Accel				
Low	0.9M/36K'	2	7	6
High	2.2M/51K'	242	230	229
7. Ground Attack Tactics	0.9M/SL	143	139	138
8. Supersonic Cruise				
Low	2.2M/39K'	242	226	225
High	2.2M/51K'	242	230	229
9. Weapon Delivery Turn				
Low	0.95M/49K'	1	7	6
High	0.95M/51K'	1	14	13
10. Flight Maneuver				
Low	0.9M/31K'	16	20	19
High	0.9M/36K'	-6	0	-1
11. Supersonic Weapon Delivery				
Low	2.2M/39K'	242	226	225
High	2.2M/51K'	242	230	229
12. Supersonic Descent				
Low	0.9M/36K'	-6	0	-1
High	2.2M/51K'	242	230	229

TABLE 3. BASEPOINT COMPOSITE MISSION FLIGHT DESIGN TEMPERATURE DATA
STANDARD DAY
CRITICAL SOLAR EFFECTS INCLUDED (CONCLUDED)

MISSION SEGMENT	REPRESENTATIVE MACH/ALT	DESIGN TEMP ±°F TOTAL TEMP	FUSELAGE	WING
13 Enroute Navigation, Return		-6	0	-1
Low	0.9M/36K'	-6	0	-1
High	0.95M/45K'	1	10	9
14. Subsonic Descent				
Low	0.39M/SL	75	79	78
High	0.9M/36K'	-6	0	-1
15. Traffic Pattern	0.39M/SL	75	79	78
16. Landing	0.2M/SL	67	69	68
17. Taxi	0M/SL	59	59	59

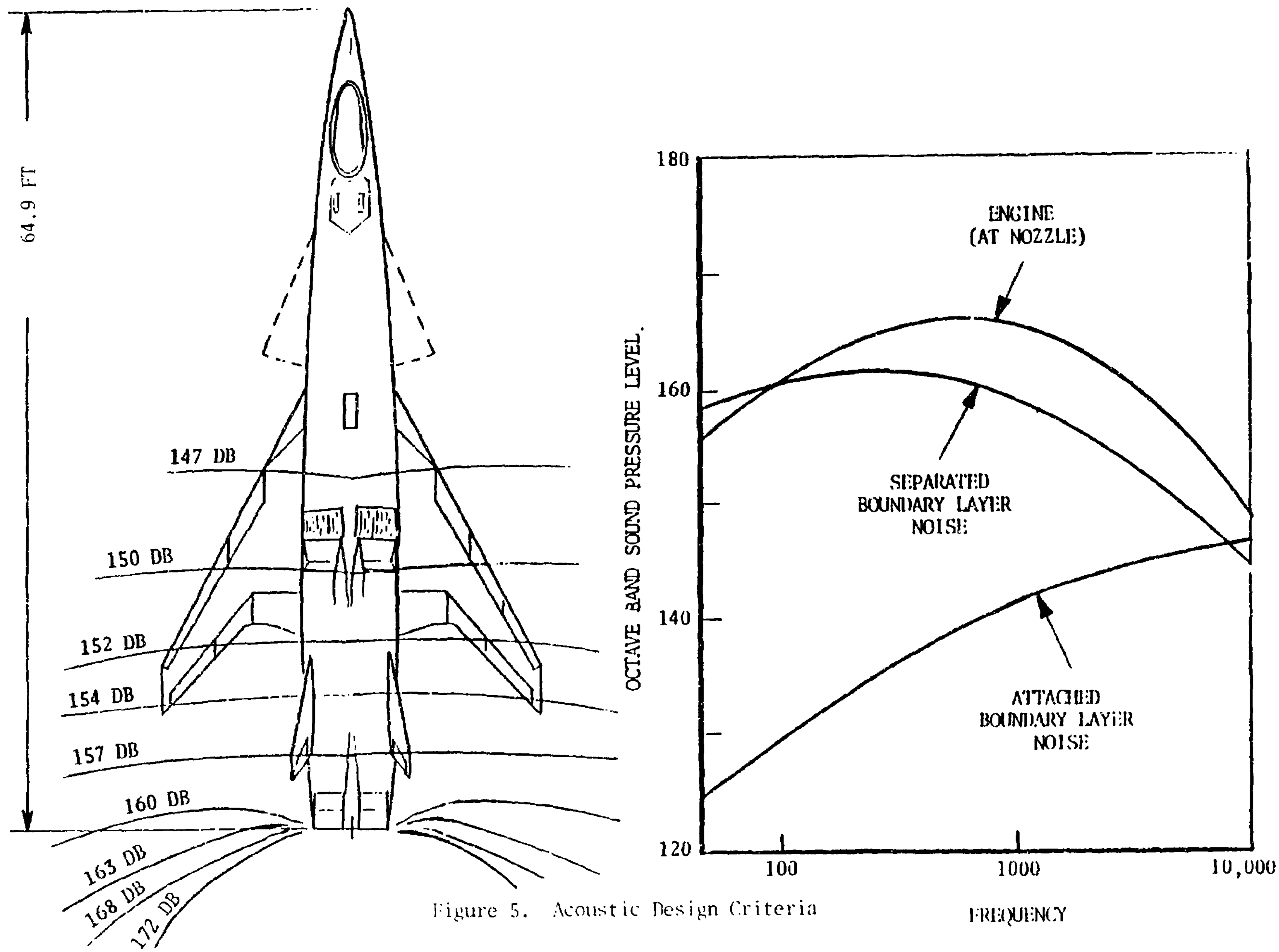


Figure 5. Acoustic Design Criteria

TABLE 4. 2.2M MISSION RELIABILITY ALLOCATION PROBABILITY OF SUCCESS (Ps)

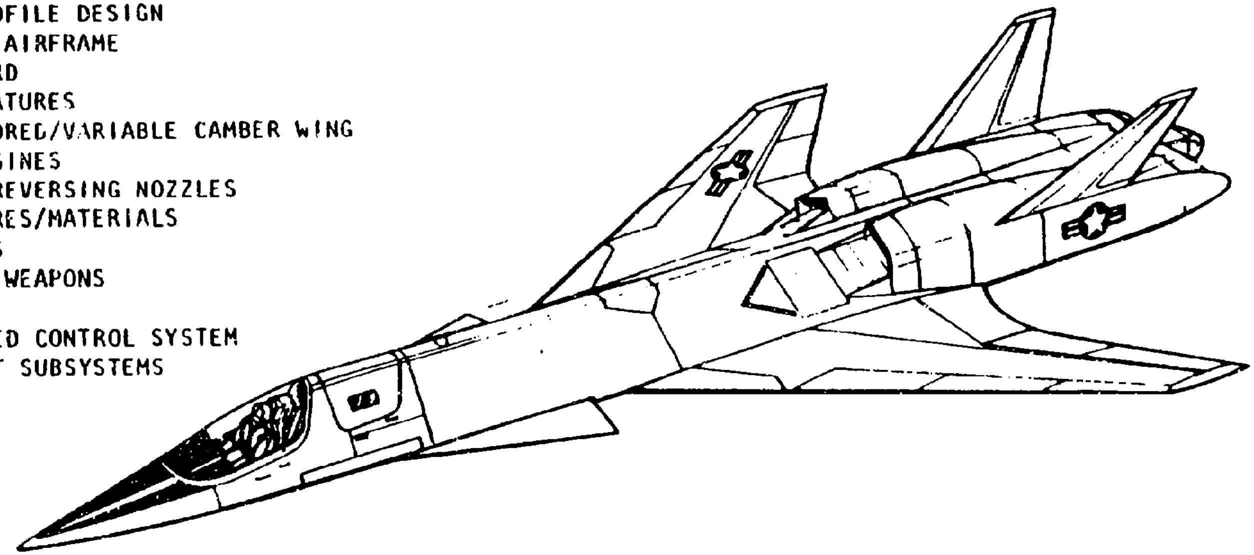
Subsystem	Baseline Ps
1. Avionics	0.9811
2. Power Plant	0.9934
3. Structure	0.9989
4. Armament	0.9983
5. Flight Controls	0.9964
6. Fuel	0.9947
7. Environmental	0.9947
8. Landing Gear	0.9976
9. Actuation	0.9976
10. Electrical	0.9984
11. Displays/Lighting	0.997
12. Auxiliary Power	0.9993
13. Crew Accommodations	0.9995
14. Other	0.998
Total	0.95

2.2 Basepoint Tactical Fighter Configuration

2.2.1 General - A representative airplane concept designed to the baseline mission requirements was established. Figure 6 illustrates the general appearance and features of the airplane. The airplane is primarily designed and sized to the high altitude 2.2 Mach number penetration mission but has significant alternate capability on the terrain following 0.9 Mach number mission. The propulsion system elements and blended wing body shaping are optimized for continuous supersonic operation. A retractable canard in combination with a vectoring 2-D nozzle provides good takeoff and landing performance with a modest installed thrust to weight ratio. The wing, canard and empennage are fabricated from advanced integral graphite/epoxy composite materials. The forward fuselage uses advanced super aluminum alloys primarily. High stress concentration areas and the aft portions of the fuselage, designed by the high ambient temperatures of the propulsion system and APU installations, are constructed of advanced superplastic formed/diffusion bonded titanium, including silicon-carbide fiber reinforced filament technology in selected areas. The wing is aeroelastically tailored and employs variable geometry features. The engine employs 3000°F turbine inlet temperatures, carbon/carbon nozzle technology and selected other advanced internal component design and material improvements. Thrust reverse capability is provided and facilitated by the 2-D nozzle configuration. The avionics installation includes a full complement of advanced technology offensive, defensive and M&TC equipment to deal with the sophisticated dual mission target and threat systems requirements. Advanced tandem mounted conformal weapons that provide standoff weapon deliveries against heavily defended targets are carried on the lower fuselage centerline. An advanced cockpit, designed around an increased seatback angle and multiple-function integrated displays to conserve space and fuselage depth, provides suitable forebody wave drag characteristics and low radar cross section (RCS) with a gold flashed canopy. The general fuselage shaping, inlet location, use of special antenna design treatments and radar absorbent materials at critical locations provides low RS characteristics.

The flight control system is characterized as a digital 3-channel fly-by-wire system with selected 4-channel portions for critical functions. The flight control will function as part of an integrated flight/fire/propulsion control system. Relaxed static stability control requirements are prescribed because it is believed that, by the 1990's, virtually all new tactical aircraft will incorporate this beneficial feature.

- MULTI-MISSION PROFILE DESIGN
- SUPERCRUISER/CCV AIRFRAME
- RETRACTABLE CANARD
- RCS REDUCTION FEATURES
- AEROELASTIC TAILORED/VARIABLE CAMBER WING
- ADVANCED 1990 ENGINES
- 2-D VECTORABLE, REVERSING NOZZLES
- ADVANCED STRUCTURES/MATERIALS
- ADVANCED AVIONICS
- ADVANCED MODULAR WEAPONS
- MODERN COCKPIT
- DIGITAL INTEGRATED CONTROL SYSTEM
- ADVANCED AIRCRAFT SUBSYSTEMS



FLIGHT CONTROL
- 3 CHANNEL DIGITAL FBW BASIC, 4 CHANNELS ON CRITICAL FLIGHT SAFETY ACTUATORS
- INTEGRATES WITH PROPULSION AND FIRE CONTROL
- RELAXED STATIC STABILITY DESIGN
- ACTUATION REDUNDANCY FOR SURVIVABILITY

*ELECTRICAL
- 115/200 VAC GENERATORS
- APU MOUNTED AUXILIARY GENERATOR
- BATTERY EMERGENCY POWER

*HYDRAULIC
- 8000 PSI, 3 INDEPENDENT SYSTEMS

*ECS
- CLOSED LOOP

*ASSUMED BASELINE SYSTEM AGAINST WHICH TRADE STUDIES WILL BE MADE

Figure 6. Basepoint Advanced Tactical Fighter (Sheet 1 of 2)

TOGW - 36,043 LB

WING AREA - 286 FT^2

ASPECT RATIO - 3.0

LEADING EDGE SWEEP - 62°

SLS THRUST/ENGINE - 9912 LB

A/G WEAPON PAYLOAD - 5000 LB

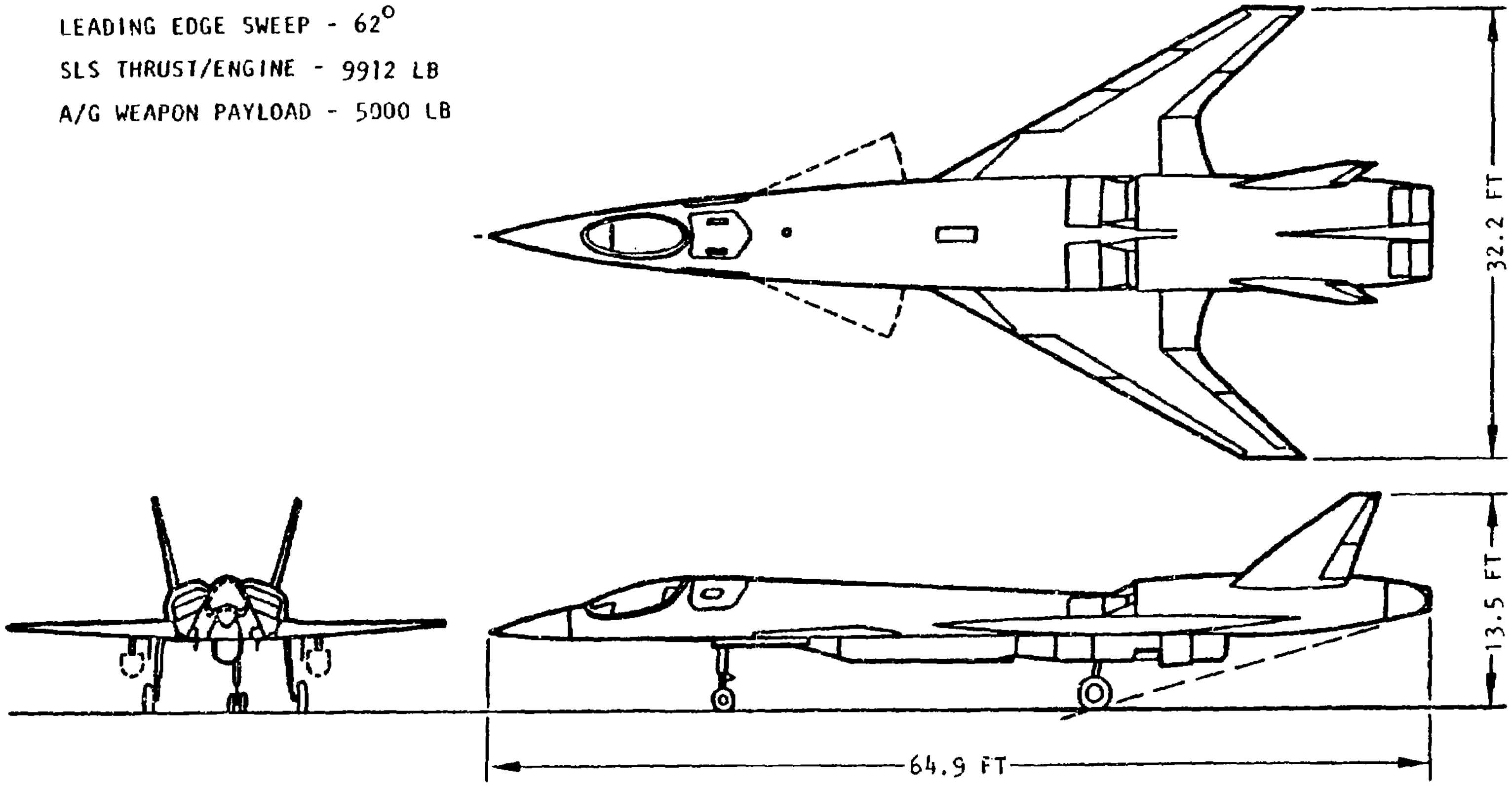


Figure 6. Basepoint Advanced Tactical Fighter (Sheet 2 of 2)

2.2.2 Power Requirements, General - In determining the power requirements the following approach was used. Aircraft power needs were generalized and established in three categories.

I Housekeeping Loads (see paragraph 4.1.4 and Appendix E)

Communications
Engine
Environmental Control
Lighting
Fuel System
Information Management
Navigation
Target Acquisition
Defensive Avionics

II Actuation Loads (see paragraph 4.1.4 and Appendix E)

Flight Controls
Utility

Armament

III Total Power Needs

Sum of Housekeeping and Actuation Loads

With the exception of environmental control, all loads under the Housekeeping Load heading were assumed to remain constant throughout the study. This was done because they represented power supplied to static (avionic black box) type of devices and thus were not considered actuation functions. All the loads under the actuation load heading plus the environmental control system loads were assumed to involve actuation functions. As such they were initially considered proper subjects for this study and ones in which the type and quantity of the power supplied would vary as different actuation methods were used during the trade study activities.

In accordance with the contract an electrical load analysis was made per MIL-E-7016 in which it was assumed that all loads were powered electrically. It was further assumed that the electrical system was 270 VDC since this type of electrical power appeared to be a likely candidate for application to 1990's aircraft. The various aircraft mission segments considered and their coding for the load analysis are as follows:

K-1 Engine start
K-2 Warm-up/take-off
K-3 Climb
K-4 Cruise
K-5 Penetration
K-6 Combat (including gun operation)
K-7 Descend
K-8 Landing
K-9 Emergency (one of the two generators failed)

Rockwell procedure for encoding this information is available in reference 36. The data was processed by a series of computer programs, and resulted in Power Source Utilization graphs such as figure 7, 8 and 9 of this report. The solid line in these figures represent the load, which is generalized into power expressed as amperes at 270 V. (The load can also be plotted as KVA or kilowatts with equal facility). The other (dotted) lines represent system capacity and interval ratings as indicated in sheet 1 of figure 7.

A separate power source utilization graph was created for each of the following Housekeeping Loads (fig. 7), actuation loads (fig. 8) and combined housekeeping plus actuation loads (fig. 9). These loads represent power supplied at the input terminals of the various output electrical devices, whether they are static black boxes or actuators. However, for the actuators their load was defined by their output (i.e. the load incident to driving compressors, powering control surfaces, etc.) and, therefore, until the type of actuator was defined, an actuator efficiency (or internal power loss) was assumed to determine the power required at the input terminals. For the purpose of this initial electrical load analysis an overall efficiency of 60% was assumed for all actuators.

The three horizontal dotted lines on each of figures 7, 8, and 9 were the interval ratings of the generator and represented;

1200 AMPS continuous capacity at 100% generation output
(two 600 AMP 270 volt D.C. generators)

1800 AMPS available for 2 minutes
(150% total generator capacity, derated to .91.5%)

2400 AMPS 5 second overload capability
(200% generator capacity, derated to 91.5%)

Note the short term overload capability of electrical power supplies compared to mechanically powered hydraulic systems whose overload capability and continuous load capacity are essentially equal.

The initial electrical load analysis was documented in Appendix A of reference 8 is included as Appendix E of this report. A breakdown of the actuation loads, which provided the basis for the "lumped" actuation load shown in reference 8, is shown in table 5.

2.2.3 Flight Control System - The aircraft is a variable stability, Control Configured Vehicle (CCV) that employs variable camber leading and trailing edge devices, a 2-D vectorable nozzle, and a variable area canard. Static longitudinal stability is set with the canard fully extended at $\delta C_m/\delta C_L = +.020$ and $M = 0.2$. Aircraft stability is varied by canard extension or retraction. This feature allows a higher trimmed CL_{MAX} with canard extended, and a reduction in wetted area for low C_L penetration and acceleration legs. Extension of the canard for transonic maneuvering allows the variable camber devices to deflect in the proper direction for both low drag and trim. Supersonic flight will require some canard extension to minimize trim drag. Table 6 summarizes

LEGEND

DERATED SYSTEM CAPACITY .... 91 1/2% OF INTERVAL RATINGS

LOAD REQUIREMENTS

SYSTEM INTERVAL RATINGS

5-SEC	200%	OF GENERATOR CAPACITY
2-MIN	150%	
CONTINUOUS	100%	

TIME INTERVAL DESIGNATIONS

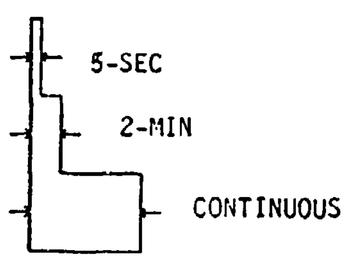


Figure 7. Power Source Utilization (Housekeeping) (Sheet 1 of 2)

ATS ELECTRIC LOAD ANALYSIS,HI ALT-MED STEALTH MISSION,0670-16-26 MAR 80 PAGE OF

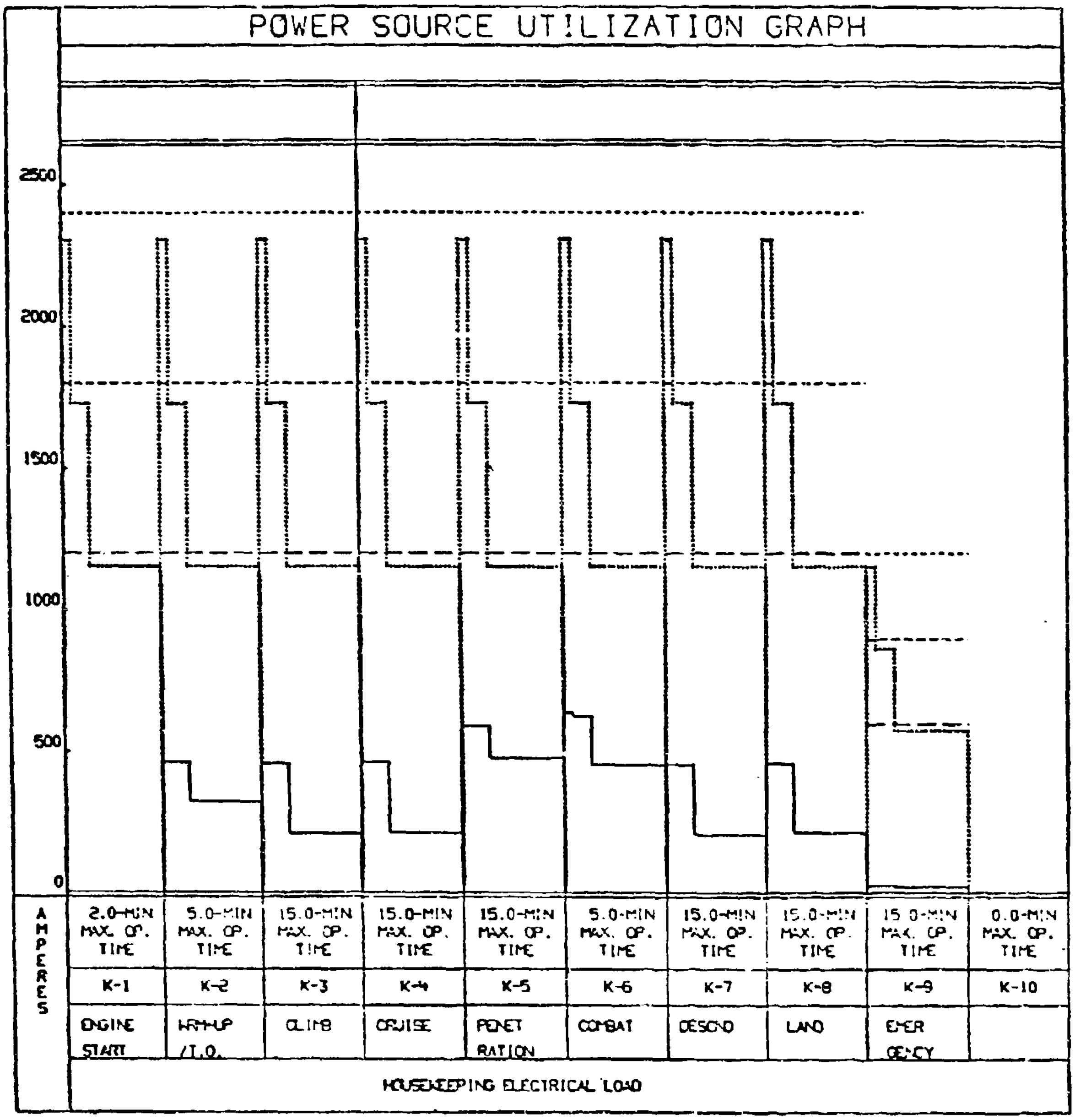


Figure 7. Power Source Utilization (Housekeeping) (Sheet 2 of 2)

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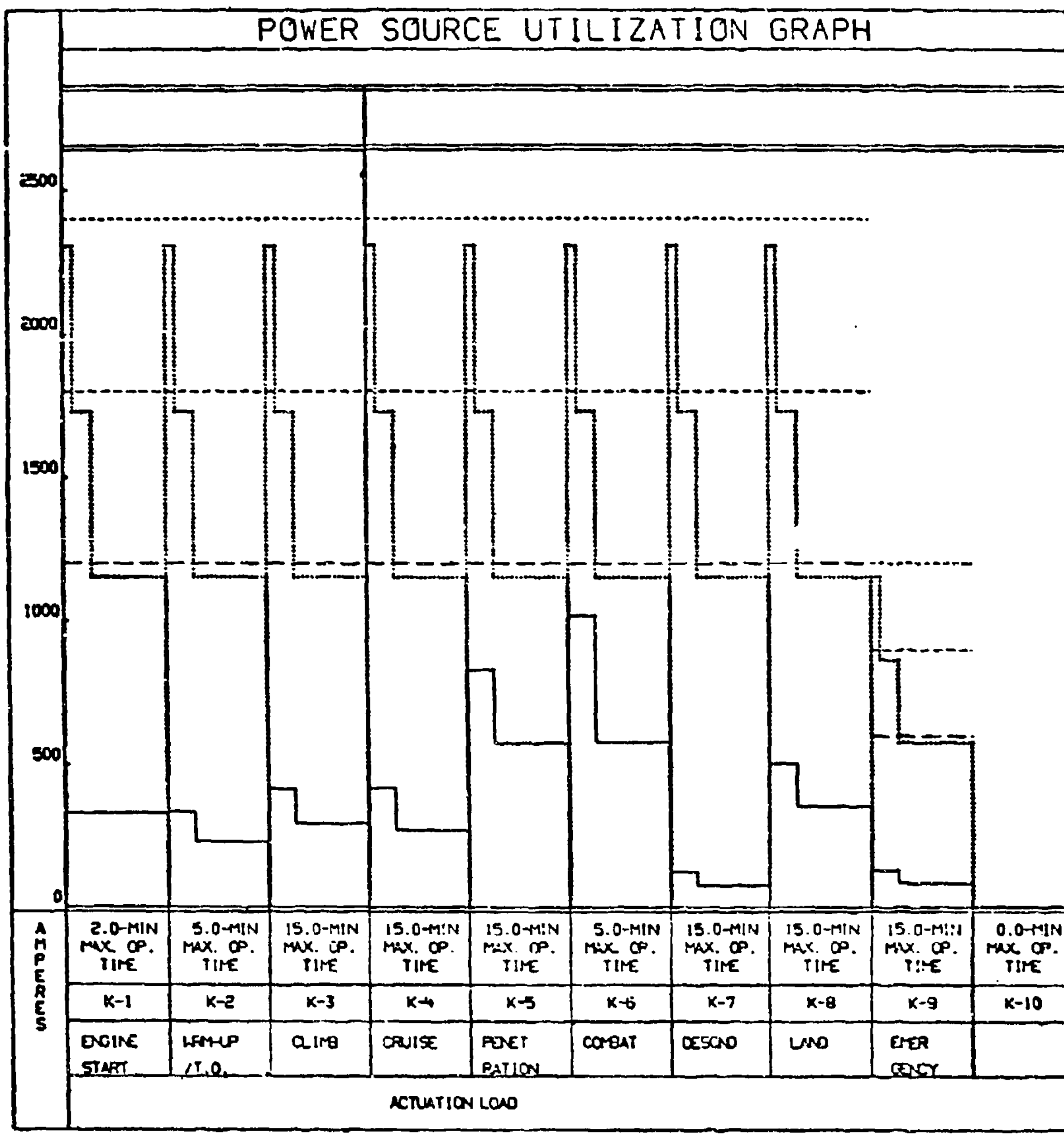


Figure 8. Power Source Utilization (Actuation)

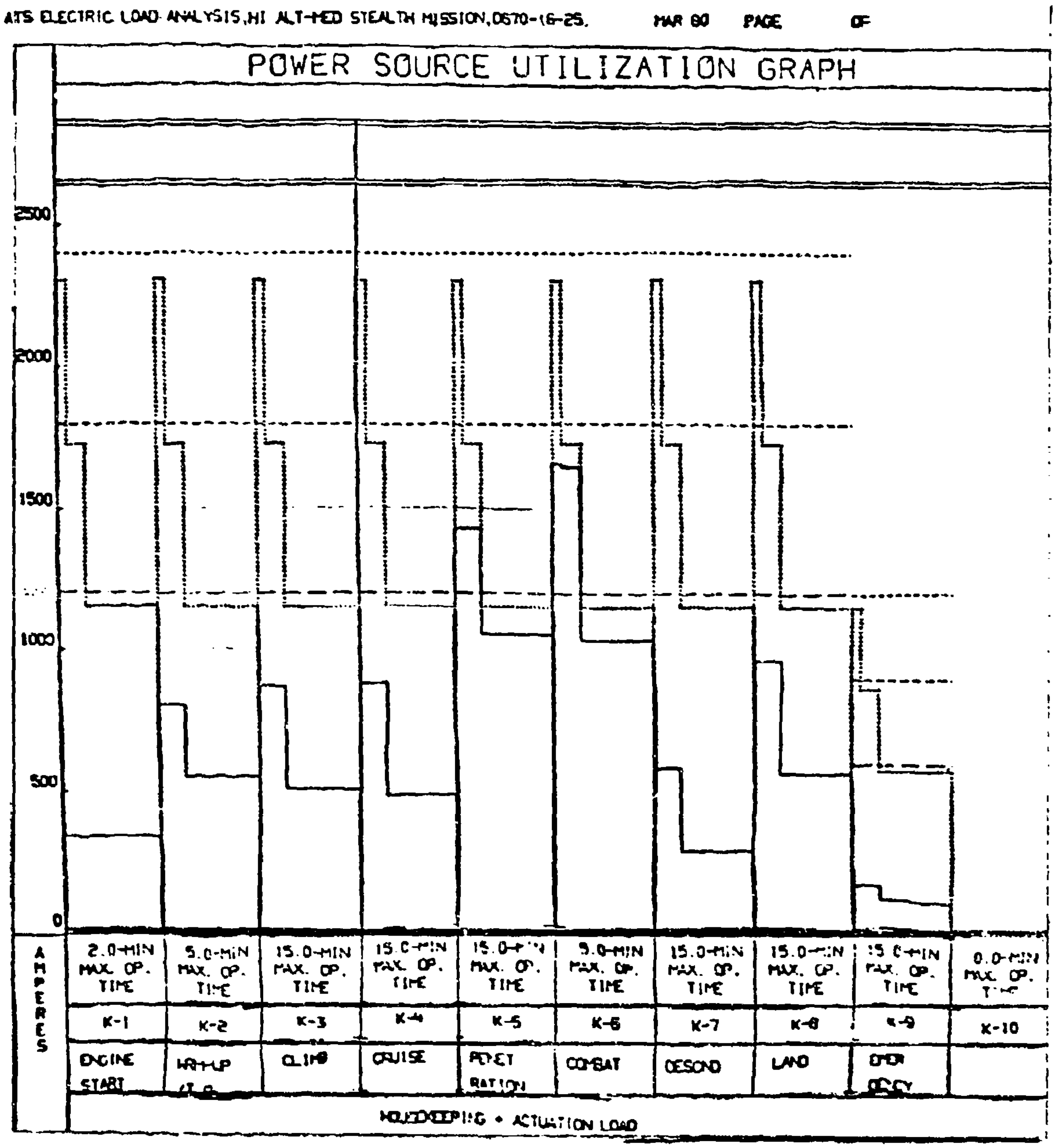


Figure 9. Power Source Utilization (Housekeeping Plus Actuation)

TABLE 5. ACTUATION LOADS (KW)

	ENGINE START		WARM-UP/T.O.		CLIMB		CRUISE		PENETRATION		COMBAT		DESCEND		LANDING		EMERGENCY	
	5 SEC	CONT	5 SEC	CONT	5 SEC	CONT	5 SEC	CONT	5 SEC	CONT	5 SEC	CONT	5 SEC	CONT	5 SEC	CONT	5 SEC	CONT
FLIGHT CONTROLS,																		
INBOARD T.E. SURFACE (2)	—	—	26.59	10.63	31.90	10.63	31.90	10.63	37.22	10.63	53.17	10.63	26.59	10.63	26.59	10.63	26.59	10.63
MIDSPAN T.E. SURFACE (2)	—	—	5.32	2.13	6.38	5.32	6.38	2.13	7.44	2.13	10.63	2.13	5.32	2.13	5.32	2.13	5.32	2.13
OUTBOARD T.E. SURFACE (2)	—	—	1.42	.57	1.70	1.42	1.70	.67	1.98	.57	2.83	.57	1.42	.57	1.42	.57	—	—
UPPER RUDDER (2)	—	—	1.37	.55	1.64	1.37	1.64	.55	1.91	.52	2.73	.55	1.37	.55	1.37	.55	1.37	.55
LOWER RUDDER (2)	—	—	1.30	.52	1.56	1.30	1.56	.52	1.82	.52	2.60	.52	1.30	.52	1.30	.52	1.30	.52
LEADING EDGE FLAP (2)	—	—	3.74	1.49	4.48	3.74	4.48	1.49	5.23	1.49	7.47	1.49	3.74	1.49	3.74	1.49	—	—
SWITCH BLADE CANARD (2)	—	—	.29	.11	.34	.29	.34	.11	.40	.11	.57	.11	.29	.11	.29	.11	—	—
ENGINE NACELLE,																		
THRUST VECTOR VANE (2)	—	—	—	—	10.82	5.41	10.82	5.41	18.04	9.02	18.04	9.02	10.62	5.41	18.04	9.02	18.04	9.02
EXTERNAL FLAP (4)	—	—	—	—	17.83	8.92	17.83	8.92	29.72	14.86	29.72	14.86	—	—	—	—	—	—
PLUG THROAT (2)	—	—	91.70	45.85	91.70	91.85	91.70	91.85	91.70	45.85	91.70	45.85	—	—	—	—	—	—
REVERSER (4)	—	—	—	—	—	—	—	—	141.28	70.64	141.28	70.64	—	—	141.28	70.64	—	—
LANDING GEAR,																		
NOSE GEAR (1)	—	—	1.40	—	—	—	—	—	—	—	—	—	—	—	1.40	—	—	—
MAIN GEAR & BRAKES (2)	—	—	2.71	.41	—	—	—	—	—	—	—	—	—	—	2.71	.41	—	—
BRAKES (1)	—	—	(1.91)	(1.91)	—	—	—	—	—	—	—	—	—	—	(1.91)	(1.91)	—	—
NOSE GEAR STEERING	—	—	.43	.10	—	—	—	—	—	—	—	—	—	—	.43	.10	—	—
ENVIRON. CONT SYS.,																		
EXHAUST DOOR EQUIP. BAY (1)	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	.01	—
RAM AIR SCOOP (3)	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	.10	—
GSD COOLING FUEL HT SINK (1)	.01	.01	.01	.01	—	—	—	—	—	—	—	—	—	—	.01	.01	—	—
SUB TOTAL	—	—	136.28	62.37	168.35	80.25	168.35	72.18	336.74	156.34	360.74	156.37	50.85	21.41	203.90	96.18	52.73	22.85
2/3 SUB TOTAL	—	—	90.84	—	112.22	—	112.22	—	224.97	—	240.47	—	33.90	—	135.92		35.15	
ARMAMENT,																		
GUN & FEED (1)					—	—	—	—	—	—	35.05	—						
TOTAL	89.50	89.60	90.84	62.37	112.22	80.25	112.22	72.18	224.97	156.34	275.52	156.37	33.90	21.41	135.92	96.18	35.15	22.85

( ) NON ADDITIVE LOADS

TABLE 6. CANARD POSITIONS

MISSION LEG	CANARD POSITION	
Takeoff and Landing	Fully Extended	100%
Subsonic Cruise	Retracted	0%
Transonic Maneuver	Fully Extended	100%
Supersonic Cruise	Half Extended	50%
Supersonic Maneuver	Fully Extended	100%
Supersonic Penetration	Retracted	0%

canard positions versus mission legs, and table 7 summarizes surfaces/ devices.

The wing trailing edges are used for pitch and roll control and, as variable camber devices with the inboard trailing edge device, the primary pitch control for longitudinal trim. The trimming function is intended to be compatible with the variable camber function of each device. A primary aerodynamic advantage of an unstable aircraft is that control deflection needed for proper camber variation is in the positive lift direction. Nose-up moments produced by an unstable aircraft as it increases angle of attack requires downward deflection of a trailing edge flap for trim and to produce the increased camber for reduced induced drag. Camber variations require a different 'neutral' setting for each flap depending on mach number. A C_{mo} change can be provided by scheduling the 2-D nozzle versus M. Therefore, the inboard trailing edge device should have the 'neutral' point scheduled versus M and a deflection versus angle of attack schedule which itself may be a function of M. This variable camber system 'automatically' produces the highest C_L available with the given planforms and control surfaces. Takeoff and landing require different flap deflections to maximize lift at zero angle of attack, C_{LO}. This second scheduling replaces conventional 'flap' settings and requires deflections of the canard and 2-D nozzle.

The midspan trailing edge flap, in addition to the inboard, is used for pitch trim and variable camber. Also, this flap is used for high speed roll control and may supplement the outboard trailing edge flap for low speed roll control. This midspan surface is used for high speed roll control in order to avoid any control reversal on the outboard surface at high q.

The outboard trailing edge is the primary low speed roll control Its function as a variable camber device is somewhat restricted in low speed flight in order not to use up control authority needed for the low speed time-to-roll requirement. Aileron 'droop' in the takeoff and landing mode can be considered useful.

The leading edge devices are primarily variable camber devices requiring scheduling versus M and angle of attack. A fixed takeoff and landing position has advantages.

The thrust vector vane is primarily a moment producing device. It optimizes flap deflections to obtain minimum drag and maximum lift. This requires scheduling as a function of M and angle of attack.

2.2.3.1 Flight Control Actuator Requirements - In arriving at the flight surface hinge moments, use was made of HiMAT generated data. The HiMAT wing planform is similar to the baseline vehicle. The ATS hinge moment data was estimated by ratioing the relative areas, chords and dynamic pressures, i.e.,

$$(HM)_{ATS} = \frac{(S)_{ATS}}{(S)_{HiMAT}} \frac{(C)_{ATS}}{(C)_{HiMAT}} \frac{(q)_{ATS}}{(q)_{HiMAT}} \times (HM)_{HiMAT}$$

TABLE 7. SUMMARY OF CONTROL SURFACES/DEVICES

Control	Travel
1. INBOARD TRAILING EDGE	-30°, $+45^\circ$
a. Primary pitch control $f(M, \alpha)$	
b. High lift device and decambering $f(M)$	
2. MIDSPAN TRAILING EDGE	-30°, $+45^\circ$
a. Pitch control $f(M, \alpha)$	
b. High lift device and decambering $f(M)$	
c. High and low speed roll control	
3. OUTBOARD TRAILING EDGE	$\pm 25^\circ$
a. Primary low speed roll control	
b. Decambering device $f(M)$	
4. CANARD	
a. Variable stability device - see table I	
5. LEADING EDGE DEVICE	
a. Variable camber device $f(M, \alpha)$	
6. THRUST VECTOR VANE	$\pm 20^\circ$
a. Pitch trim $f(M, \alpha)$	
7. RUDDERS	$\pm 25^\circ$
a. Directional control	
b. Speed brakes	

The ATS rudder hinge moments were computed from the ATS midspan trailing edge flaps (the HiMAT has all movable flip-flop verticals). The ATS canard loads were estimated from the canard lift force calculated at maximum design load factor (7.3 g's) at maximum dynamic pressure (1750 PSF). The leading edge design loads were based on an assumed 10 psi maximum pressure acting on an arm equal to one half the flap chord. The thrust vector actuator loads were taken from NASA CR 135252 which used an exhaust nozzle and engine similar to the baseline ATS.

Three of the trailing edge control surfaces were examined for flutter stiffness. Estimates of frequency were derived from empirical data. For the inboard flap, the frequency was estimated to be 47.9 hz, 95.8 hz for the upper rudder. The mach number used was 1.25 while for the aileron the Mach number was .95. The stiffness computed from the empirical data was increased by 50 percent to account for surface wind up. The backup structure was thus assumed to have equal stiffness with the actuator. The stiffness requirements for the three surfaces considered are given in table 8 which presents the performance requirements for the actuation systems for the various flight control surfaces/devices identified for the baseline vehicle.

The actuation requirements listed in table 8 need not be satisfied simultaneously. For a worst case analysis, the simultaneous requirements are: 1) 70 percent for inboard and midspan flaps, 2) 50 percent for the rudders, and 3) 100 percent for the ailerons. The other flight control actuation devices can be assumed to be stationary during this simultaneous demand.

As indicated earlier, the flight control system is a fly-by-wire design with fail-operate-twice capabilities for the flight critical surfaces. These surfaces are at least triple redundant, electrically and mechanically.

For the other surfaces/devices, triple redundancy is provided electrically but only dual redundancy mechanically. For the flight critical surfaces full actuation performance is required after any two electrical or mechanical failures or any combination of single electrical and mechanical failures. Any further failure will not produce a hardover deflection. For the non-flight critical surfaces/devices, full actuation performance is required after any single failure. After any further failure, the surface/device will be capable of being recentered and locked.

The actuator position loop closures and the monitoring/switching provisions will be included in the actuation system. Electrical digital signal inputs, when used, will be 5 volts.

Table 8 specifies frequency response in two methods. Method one establishes actuator output response requirements for a fixed input signal of increasing frequency, and method two establishes output response requirements for fixed amplitude output at increasing frequency response. Loads are considered to be essentially inertia. For purposes of this study linear load variation with stroke/rotation will be considered.

2.2.3.2 <u>Utility Actuation</u> - Table 9 summarizes utility actuation characteristics.

TABLE 8. FLIGHT CONTROL ACTUATION REQUIREMENTS

FUNCTION	TRAVEL	LOAD CHARACTERISTICS (3): NO LOAD (MAXIMUM) RATE	LOAD CHARACTERISTICS (3): DESIGN LOAD AND RATE	LOAD CHARACTERISTICS (3): MAX. LOAD AND/OR STALL	NO. OF SURFACES	POWER PER SURFACE AT DESIGN (3) LOAD & RATE	DUTY CYCLE	(1) RESOLUTION (R)	TOTAL ERROR SUM (E) OF HYSTERISIS & RESOL.	CLOSED (1) LOOP (2) FREQUENCY RESPONSE	STIFFNESS	FAILURE MODE / INERTIA (I)	AMPLITUDE INPUT (1)	AMPLITUDE OUTPUT (2)
INBOARD FLAP	-30° TEU +45° TED	100° PER SEC.	161,664 IN-LB AT 50°/SEC.	215,552 IN-LB.	2	15.95 KW	PER MIL-F-9490C ¶ 3.3.9.1 AND 3.3.9.2	R ≤ 0.05% FULL STROKE	E ≤ 0.2% FULL STROKE	AMPL +2DB -3DB ϕ -30° MAX AT 3 HZ	1.82×10⁶ IN-LB/RAD	FO² FS I = 19330 LB-IN²	± 0.25°	± 1.0°
MIDSPAN FLAP	-30° TEU +45° TED	100° PER SEC.	32,344 IN-LB AT 50°/SEC	49,125 IN-LB.	2	3.19 KW		R ≤ 0.05% FULL STROKE	E ≤ 0.2% FULL STROKE	AMPL +2DB -3DB ϕ -30° MAX AT 3 HZ	—	FO² FS	± 0.25°	± 1.0°
AILERON	± 25°	50° PER SEC.	21,453 IN-LB AT 20°/SEC	25,883 IN-LB	2	0.85 KW		R ≤ 0.2% FULL STROKE	E ≤ 0.5% FULL STROKE	AMPL +2DB -3DB ϕ -45° MAX AT 2 HZ	4.57×10⁵ IN-LB/RAD	FO FS I = 914 LB-IN²	± 0.6°	± 1.0°
UPPER RUDDER	± 20°	40° PER SEC.	13,871 IN-LB AT 30°/SEC.	18,495 IN-LB	2	0.82 KW		R ≤ 0.2% FULL STROKE	E ≤ 0.5% FULL STROKE	AMPL +2DB -3DB ϕ -30° MAX AT 3 HZ	2.58×10⁶ IN-LB/RAD	FO² FS I = 687 LB-IN²	± 0.5°	± 1.0
LOWER RUDDER & SPEED BRK	± 20°	40° PER SEC.	19,816 IN-LB AT 20°/SEC.	26,421 IN-LB	2	0.78 KW		R ≤ 0.2% FULL STROKE	E ≤ 0.5% FULL STROKE	AMPL +2DB -3DB ϕ -30° MAX AT 3 HZ	—	FO² FS	± 0.5°	± 1.0°
LEADING EDGE FLAP	20° LED	5° PER SEC.	473,032 IN-LB AT 2.4°/SEC	520,335 IN-LB	2	2.24 KW		R ≤ 0.5% FULL STROKE	E ≤ 1.0% FULL STROKE	AMPL +2DB -3DB ϕ -45° MAX AT 2 HZ	—	FO FS	0.6° LED	1.0° LED
SWITCH BLADE CANARD	20° EXT 15.36 IN.	3° PER SEC. 2.3 IN/SEC	2000 LB. AT 1.0°/SEC 0.77 IN/SEC	2400 LB.	2	0.17 KW		R ≤ 0.5% FULL STROKE	E ≤ 1.0% FULL STROKE	AMPL +2DB -3DB ϕ -45° MAX AT 2 HZ	—	FO FS	0.6° EXT. 0.46 IN	1.0° EXT. 0.77 IN.
THRUST VECTOR VANE	± 20° ± 2.10 IN.	3° PER SEC. 0.31 IN/SEC	64,017 LB AT 1.0°/SEC. 0.10 IN/SEC.	75,314 LB	2	0.72 KW		R ≤ 0.4% FULL STROKE	E ≤ 1.0% FULL STROKE	AMPL +2DB -3DB ϕ -45° MAX AT 2 HZ	—	FO FS	± 1.0° ± 0.10 IN.	± 1.0° ± 0.10 IN.

TEU = TRAILING EDGE UP
TED = TRAILING EDGE DOWN
LED = LEADING EDGE DOWN
EXT = EXTEND

(1) INPUT SIGNAL AMPLITUDE CONSTANT AT 6 TIMES RESOLUTION (METHOD #1)
(2) INPUT SIGNAL AMPLITUDE VARIED TO MAINTAIN OUTPUT AMPLITUDE CONSTANT (METHOD #2)
(3) AT CONTROL SURFACE

FO = FAIL OPERATE
FO² = FAIL OPERATE, FAIL OPERATE
FS = FAIL SAFE
DB = DECIBELS

TABLE 9. BASELINE UTILITY ACTUATOR REQUIREMENTS

	FUNCTION	TRAVEL	LOAD CHARACTERISTICS STALL	OPERATING	LOAD/RATE	POWER AT OPERATING LOAD-RATE ①	POWER AT INTERFACE ②	FAILURE RESPONSE	NO. ACT REQD. /AC	ACTUATOR TYPE ③ HYD	ELEC	COMMENTS
ENGINE	EXTERNAL FLAP	10.0 IN	8600 LB	7310 LB	5.00 IN/SEC	4.13 KW (5.54 HP)	—	FAIL SAFE (FS)	4	HLA	MBS	MODULATES
	PLUG THROAT	9.25 IN	57,400 LB	46,790 LB	4.63 IN/SEC	25.52 KW (34.21 HP)	28.67 KW (38.43 HP)	(FS)	2	MPT	MPT	POWER TRAIN η = .96 BALL SCREW x .95 RT ∠ RED. GEARBOX x .98 LINKAGE = .89 10,000 RPM AT INTERFACE, MODULATES
	THRUST REVERSER	12.32 IN	16,600 LB	14,110 LB	12.32 IN/SEC	16.89 KW (22.64 HP)	19.64 KW (26.33 HP)	(FS)	2	MPT	MPT	2 POSITION - POWER TRAIN η = .99 INPUT x .95 GB#1 x .95 GB#2 x .96 BALL SCREW = .86 - 10,000 RPM INPUT
LANDING GEAR	NOSE GEAR	5.00 IN	—	14,900 LB	0.83 IN/SEC	1.40 KW (1.88 HP)	—	(FS) OPERABLE AFTER ONE POWER FAILURE	1	HLA	MBS	
	MAIN GEAR	6.00 IN	—	24,000 LB	1.00 IN/SEC	2.71 KW (3.63 HP)	—		2	HLA	MBS	
	NOSE GEAR STEERING	±101°	—	18,000 IN-LB	12°/SEC (2 RPM)	0.43 KW (0.58 HP)	0.48 KW (0.64 HP)	(FS) TRAIL	1	MPT	MPT	GEAR TRAIN η = .90 MODULATES 20,000 RPM AT INTERFACE
	MAIN GEAR BRAKES	0.10 IN	75,282 LB	33,800 LB	0.50 IN/SEC	1.91 KW (2.56 HP)	3.18 KW (4.27 HP)	(FS)	2	MP	MP	HYD. CIRCUIT η = .87 MOTOR x .86 PUMP x .80 VALVING = .60 12,000 RPM AT INTERFACE
ENVIR. CONT. SYS.	RIGHT RAM AIR SCOOP	2.52 IN	—	4100 LB	2.0 IN/SEC	0.93 KW (1.25 HP)	—	(FS)	1	HLA	MBS	
	LEFT RAM AIR SCOOP	1.00 IN	± 1200 LB	±650 LB	0.15 IN/SEC	0.01 KW (0.013 HP)	—	(FS)	1	HLA	MBS	
	EXH. DOOR EQUIP. BAY	2.66 IN	1300 LB	900 LB	0.09 IN/SEC	0.01 KW (0.013 HP)	—	(FS)	1	HLA	MBS	
	GRD. COOLING FUEL HT. SINK	90°	150 IN-LB	± 90 IN-LB	18°/SEC	.003 KW (.004 HP)	.006 KW (.008 HP)	(FS)	1	MBS	MBS	ALWAYS ELECTRICALLY ACTUATED - 25,000 RPM AT INTERFACE - TOO SMALL A POWER REQ FOR HYDRAULICS
MISC. UTILITY FUNCT.	ARMAMENT	G.E. 30MM CANNON & AMMUNITION FEED SYSTEM < 5.0 SEC. BURSTS - NO CONTINUOUS FIRING				35.05 KW (47.00 HP)	—	(FS)	1	M	M	PROGRAMMED FIRING CYCLE WITH REVERSE ROTATION AT END OF CYCLE TO RECOUP UNFIRED SHELLS
	ENGINE START	MAX. STARTING TIME PER ENGINE = 40 SEC				89.49 KW (120 HP)	—	(FS)	2	ATS	GEN	
	REFUEL RECEPTACLE	STANDARD (UARRSI) REFUELING RECEPTACLE (3000 PSI HYDRAULIC)				0.54 KW (0.72 HP)	0.83 KW (1.11 HP)	(FS)	1	MP	MP	HYD CIRCUIT η = .85 PUMP x .80 VALVE x .95 ACTUATOR = .65 20,000 RPM AT INTERFACE, 2 POSITION

① POWER PER ACTUATOR DELIVERED AT THE SURFACE OR FUNCTION BEING OPERATED

② POWER DELIVERED AT A MOTOR TO PUMP, OR MOTOR TO GEAR TRAIN, INTERFACE IN WHICH EVERYTHING DOWNSTREAM, BETWEEN IT AND THE FUNCTION, IS UNCHANGED WHETHER THE POWER SYSTEM IS HYDRAULIC OR ELECTRICAL.

③ ATS = AIR TURBINE STARTER　GEN = GENERATOR
MP = MOTOR-PUMP　M = MOTOR
MBS = MOTOR-BALL SCREW
HLA = HYDRAULIC LINEAR ACT.
MPT = MOTOR-POWER TRAIN (MAY INCLUDE SHAFTING, BALLSCREWS, & GEARS)

2.2.4 Environmental Control System - The environmental control system (ECS) in the aircraft provides proper conditions for crew, avionics, and missiles. The crew requires cockpit pressurization, heating, cooling, and ventilation in addition to windshield and canopy defog, windshield anti-ice and rain removal. The missiles require air for pre free flight conditioning and the avionics requires the removal of self generated heat. A breakdown of the maximum continuous heat loads imposed on the ECS are shown in Table 10.

In order to provide the necessary data for the initial electrical load analysis (see Appendix E) an ECS system, felt to be representative of the type which might result from this study, was hypothesized. The resulting system is shown in figure 10. The system derives most of its input power from shaft inputs at its various compressors, pumps, and blowers. These shaft inputs could be supplied either directly from the engine or via electric or hydraulic motors. A small amount of power (<10%) was derived from bleed air to pressurize the cockpit and provide makeup air.

Figure 10 shows that the maximum continuous power required per the electrical load analysis was 80.17 K.W. This loading was considered to exist during penetration and combat. During all other mission segments the maximum continuous load was considered to be 44.22 K.W. The five second peak loads were 136% of these values and were 109.03 K.W. and 60.14 K.W., respectively. Figure 10 also shows that heat is rejected to fuel and lists some of the salient features of the system.

2.2.5 Aircraft Configuration - The inboard profile for the aircraft selected as the baseline for the study is shown in figure 22. The figure also lists most of the major subsystem components used in the aircraft as it was originally conceived. Further discussion of this configuration will be found in paragraph 4.1.3.

2.2.6 Armament Subsystem - The gun carried by the aircraft was a GE430 four barrel, 30 mm Gatling type. The gun required 22 hp steady state while the linear linkless feed system (LLFS) required 25 hp. The total gun requirement was 47 hp.

Air-to-air (ATA) missiles were semisubmerged on the underside of the fuselage. The missiles were launched by forcible ejection; jettison mode was provided for emergency release. Electrical, hydraulic and environmental air conditioning lines run through the fuselage to connect with the missile for preflight conditioning and release.

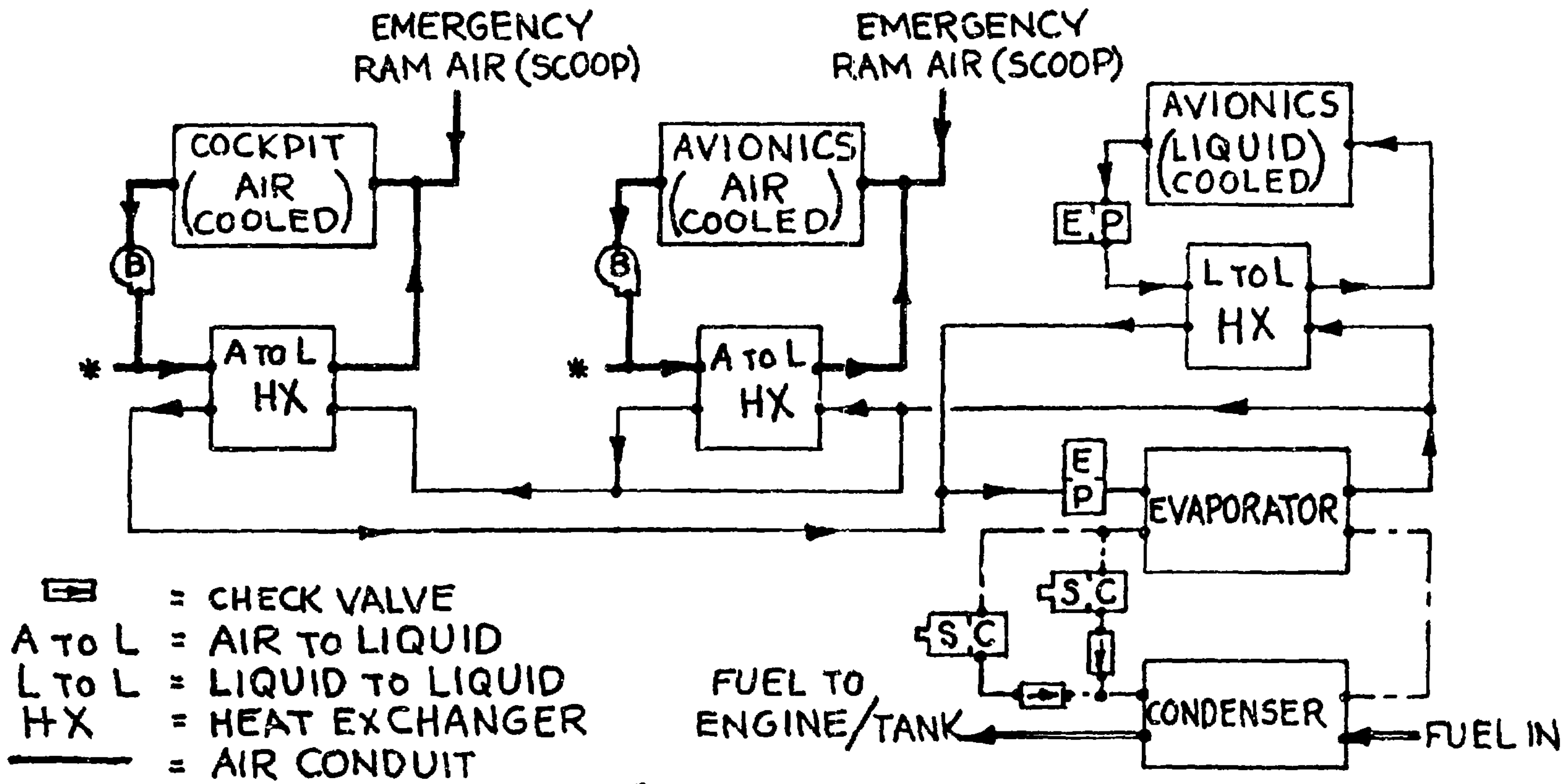
Design mission air-to-ground weapons were conformal and were carried in tandem. Standard ejectors supported the weapons and provided forcible ejection for safe separation under all aircraft flight conditions.

2.2.7 Engine Starting Loads - Figure 11 shows the starting characteristics for the F404 GE 400 Engine. This was a 16000 lb S.L.S. thrust engine which required 125 starting horsepower on a Standard Day for a 35 second start. The F-18 Sundstrand Starter ATM08 used on the engine could generate 167 HP on a standard day.

TABLE 10. ECS HEAT LOAD

Subsystem	Heat Load KW	
Cockpit		3.51
Avionics		
Armament	2.11	
Communications	1.93	
Engine	.79	
Information Mgmt. Sys.	2.62	
Navigation	.59	
Target Acquisition	14.67	
Defensive Subsystem	21.50	
A/V Electrical System	.50	
	44.76	
25% Growth	11.19	
	55.95	55.95
		59.46
Cockpit Pressurization		4.27
		63.73
*Composite circulation loop efficiences		.935
Required output of pumps & blowers		68.18

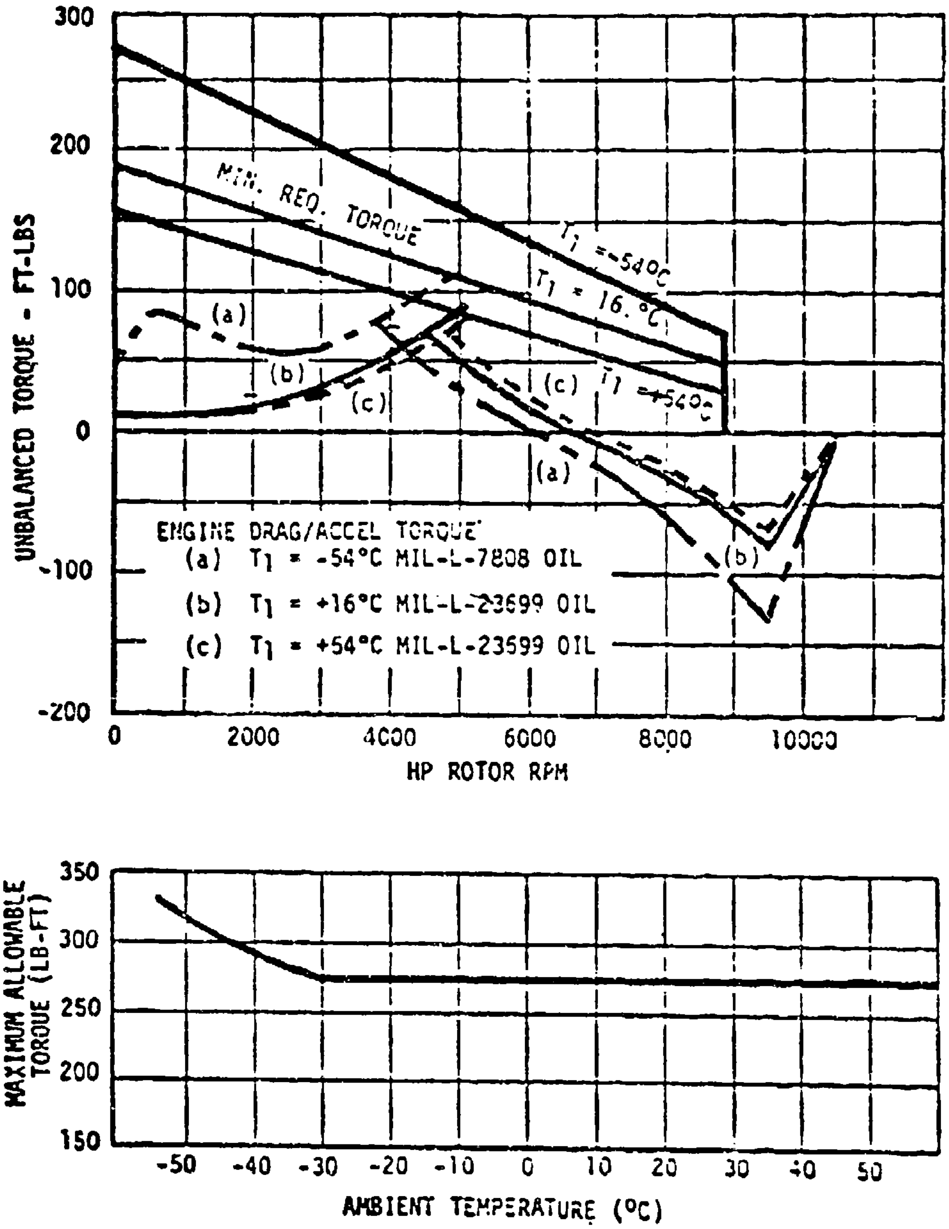
* 0.93 Freon Loop
0.98 Air Loops
0.95 Liquid Loops



= CHECK VALVE
A TO L = AIR TO LIQUID
L TO L = LIQUID TO LIQUID
HX = HEAT EXCHANGER
= AIR CONDUIT
= LIQUID COOLANT LINE
= FREON CIRCUIT

	NO. REQ/AC	OUTPUT LOAD/ITEM	INPUT LOAD/ITEM	TOTAL INPUT LOAD/AC
EP = ELECT. MOTOR DRIVEN PUMP	2	.74 KW	.87 KW	1.74 KW
SC = SHAFT DRIVEN COMPRESSOR	2	30.56 KW	35.95 KW	71.90 KW
B = ELECT. MOTOR DRIVEN BLOWER	2	2.74 KW	3.22 KW	6.43 KW
TEMPERATURE CONTROL	1			.10 KW
TOTAL SHAFT POWER REQD.				80.17 KW
* PRESSURIZATION (BLEED AIR)				4.27 KW

Figure 10. Aircraft I & Aircraft II ECS System



NOTES:

(1) No customer air bleed during starting.

(2) Power extraction permitted between starter cutout and ground idle, up to 34 HP providing starter cut out is at least 5900 rpm ± 200 rpm.

(3) Minimum engine firing speed 3900 rpm.

Figure 11. Starting Torque and Speed Requirements
(Sea Level Static Conditions)

The ATS engine SLS thrust is 9912 lbs and, proportionately scaling starting H.P., would call for slightly more than 105 H.P. However, accessory drag will not decrease proportionately, therefore, for purposes of this study, starting horsepower was established at 120 H.P. (89.5 K.W.).

2.3 Historical Review

2.3.1 Data Revisions and Additions - Subsequent to the definition of the baseline aircraft, as discussed in paragraphs 2.1 an 2.2 certain baseline aircraft requirements data were revised or expanded and clarified. These data items are discussed in the following paragraphs:

2.3.1.1 Ground Cooling Fuel Heat Sink Door - The ground cooling fuel heat sink was originally included but was subsequently eliminated as an item in the trade study. This arose from the fact that the power requirement was so small that it would not have been practical to perform the function hydraulically. The smallest motor it is feasible to manufacture delivers approximately 700 times the power required for this application. A linear actuator sized to perform the function would have been so small that the port bosses necessary for providing extend and retract pressure would have represented more that 50% of the total volume of the actuator. These small power functions have historically been, and will continue to be operated electrically. Therefore, it was assumed that this function would be performed electrically on both baseline aircraft. This is shown in the "comments" column of Table 9.

2.3.1.2 Canopy - The canopy was also originally included and later eliminated as an actuation function. Extensive study indicated that the canopies should be opened and closed manually and thus should not be a utility actuation function. The decision to return to manual operation came during a rather detailed analysis, in terms of weight and reliability, of the actuation system (electrical or hydraulic) necessary to open or close the canopies. This analysis considered the requirements with the power supply system failed or during ground maintenance with no power on the power supply system. To meet these requirements the system became very complex and heavy, particularly when meeting the range of conditions which could reasonably be expected during normal operation. Complicated as it was such an actuation system would have been necessary had the canopy system consisted of one large canopy rather than two separate smaller canopies. However, since the canopy system was broken into two canopy units the airloads acting on each unit and the weight of each was considerably less. Thru the use of counter balancing, in the form of torsion bars or gas springs, the static deadweight loads could be nearly eliminated and each canopy unit could be handled easily under all but the severest gust load conditions. Since each canopy unit was automatically locked open (manual unlock to close) and manually locked closed, all functions were manual. Even though opening the canopy against the most severe adverse wind loads would take considerable physical effort, the fact that the unit locked open automatically meant that this level of effort had to be sustained for only a short period of time.

Manual operation of the canopy offered many benefits to the aircraft a few of which were as follows:

A. Reduced aircraft weight

B. Simplified power distirbution and utilization system whether electrical or hydraulic.

C. Improved aircraft reliability under normal operating conditions.

D. Improved maintainability arising from both the simplified actuation system and from improved access to the cockpit during routine maintenance (i.e. the canopy can be opened and closed an indefinite number of times without the need for batteries, accumulators or ground power.

E. Improved emergency access or egress.

2.3.1.3 Utility Engine Actuation Functions - Two of the three major power users during the critical combat phase (i.e. the operational phase which determined power generation system size) were the plug-throat and the thrust reverser. Figure 12 shows the actuation mechanism which operates these functions plus the external flap and thrust vector vane. It will be noted that the plug-throat, the thrust reverser and thrust vector vane were all operated via various power trains driven by actuation devices mounted in the sidewall at the ends of the rectangular engine exhaust duct. The plug-throat was driven by a shaft running down the center of the intermediate sidewall cross tie while the thrust reverser was powered by a shaft mounted on the front face of the aft side wall cross tie. In this way the motors or linear actuator (i.e. the power transducers) could be mounted in the relatively cool sidewall area while the shafts, gearboxes and ballscrews, which were more temperature tolerant, were mounted in the hot (400°F with cooling-1000°F without) nozzle area.

Originally it had been hoped that the thrust reverser power transducing devices, for the electrical system or the hydraulic system or both, could be mounted in the hot area. This would have made possible the elimination of all, or a large part of, the existing power train in the hot area and thus would have reduced complexity and weight. Towards this end an envelope was given in Figure 2-12 defining thrust reverser installation requirements assuming all actuation devices were in the hot area. Subsequent studies however, indicated that the ductwork necessary for, and the induced drag increase associated with, attempting to cool the envelope area down to 400°F max would offset most of the potential weight savings and this approach was dropped. It was finally assumed that the thrust reverser and the plug-throat would both be operated by 10,000 RPM motors (electric or hydraulic) mounted in the relatively cool engine exhaust nozzle side wall.

2.3.1.4 Environmental Control System - As previously pointed out (paragraph 2.2.4) an ECS System was hypothesized to make possible the determination of ECS loads in case the ECS system became an integral part of the trade study. It was also pointed out that the system derived most of its input power from shaft inputs at its various compressors, pumps and blowers. These shaft inputs could have been supplied either directly from the engine or via electric hydraulic motors. A small amount of power (4.27 KW) was derived from bleed air to pressurize the cockpit and provide makeup air. The balance was supplied in the form of shaft power inputs.

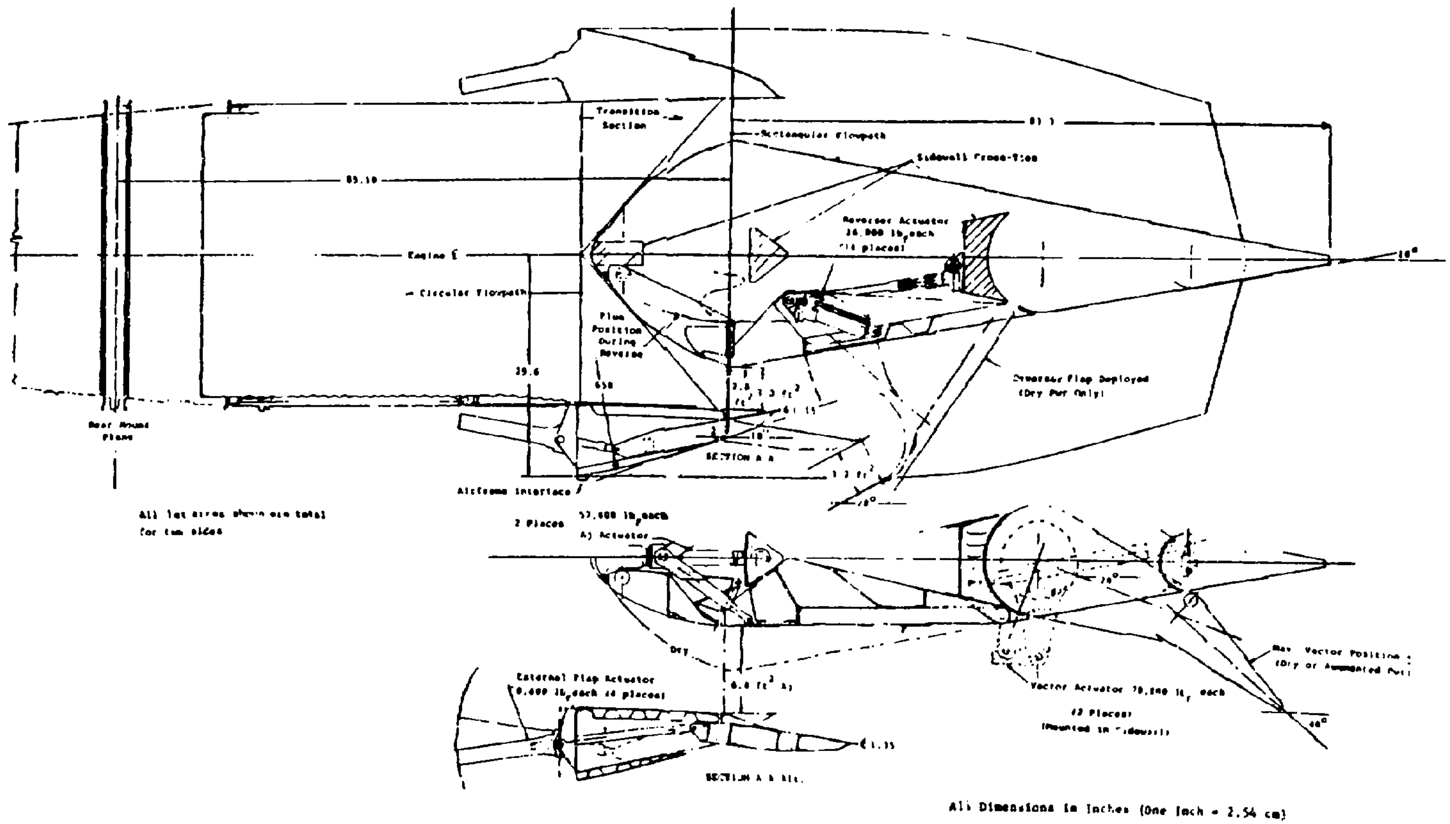


Figure 12 Engine Actuation Functions

Figure 10 shows that the maximum continuous power required was 80.17 KW of which 9.27 KW must be carried by the electrical system. This loading was considered to exist during penetration and combat. During all other mission segments the maximum continuous load was 44.22 KW.

After considerable study it was determined that the ECS system for the aircraft which were to be traded (i.e., aircraft I and aircraft II, see paragraph 2.4.6 for definition) could, and should, be identical and thus should be dropped as an item in the trade study. The factors which led to this determination were as follows.

1. The heat loads seen by the ECS system in both aircraft were identical. This was true because the only heat load variations between aircraft were those generated by the actuation systems and all actuation systems, whether electro mechanical, hydraulic or integrated actuator package type, reject heat to their surrounding ambient (air or fuel) and not to the ECS system. The heat load represented by cockpit, avionics (air) and avionics (liquid), Figure 10, was unchanged for all study aircraft. Hence, the blowers, pumps and heat exchangers servicing these heat loads were unchanged. In effect the ECS dropped out as a trade study item except for the impact of its small load requirement on the two different types of electrical power generation systems used in each aircraft. The only additional factor which was considered was the diffences between actuation approaches with reference to their heat load impact on the fuel. For those approaches which tended to overheat the fuel, a weight/cost/reliability penalty was assessed in term of increased condenser size and/or the addition of auxiliary fuel cooling equipment.

2. The compressors were shaft driven. This approach was selected because direct shaft power extraction was much more efficient than having an intermediate hydraulic or electrical transmission link (97% shaft vs 72% with electrical or hydraulic). Since the compressor represented by far the major load imposed on the propulsion system by the ECS system, this increased efficiency was greatly to be desired. The desireability of direct shaft power extraction was further enhanced by the fact that the compressor should be close to the condenser, which must be in the final fuel inlet line to the engine, which, in turn, was adjacent to the AMAD (see Figures 17 and 22).

3. All blowers and circulating pumps were electric motor driven in both aircraft I and aircraft II. This decision was based on historic data and B-1 aircraft experience. Historically, on most aircraft, ECS blowers and ECS coolant circulating pumps have been electric motor driven and the results have been generally very satisfactory. In contrast, on the B-1 aircraft, several blowers and coolant pumps were hydraulic motor driven. The results have not been satisfactory. The motor ripple frequency interacting with vibrational frequencies generated by the coolant pumps and blowers have caused erratic unpredictable premature failures. This fact, plus the fact that the individual pump loads (.87 KW) were so small that they are well below the power capability of the smallest 8000 psi hydraulic motor it will be practical to manufacture in the 1990+ time period (i.e. practical minimum 4.0 KW), indicated quite clearly that these motors should be electric. The blower motor loads were also marginal but the deciding factor in this instance was the fact that the blowers were in pressurized compartments (avionics and cockpit) where it was desireable not to have fluid leaks.

2.3.1.5 Redundancy Definitions and Percent Output Load Requirements - A table was prepared (Table 11) to define what constituted fail safe for the various flight control actuation functions. The same table also more accurately defined the actual output load requirements for the various flight control functions in the presence of failures. This definition stated in essence that after first failure all actuation functions must retain 100% load capability. This was in line with the overall air vehicle requirement that the aircraft be able to complete its mission after any single power generation and/or distribution system failure (hydraulic or electric). The refined definition also conformed with air vehicle requirement by stating that after any second failure, each critical actuation function must retain sufficient power capability to allow the aircraft to recover from any manuver and return to base. For this purpose the residual power capability of each actuation function must range from 50 to 70%. The amplified and redefined load requirements are shown in Table 2-11. It can be seen in this table that the rudders required only 50% capability after two failures while the inboard (and midspan) flaps required 70%.

2.3.1.6 Installation Envelopes - In order to more accurately define the problem of installing actuation devices in the ATS aircraft three installation envelopes were provided as additional requirements data. These envelopes were for the outboard trailing edge (Figure 13), the inboard trailing edge (Figure 14), and the thrust reverser (Figure 12). The outboard trailing edge represented the smallest chordal thickness application, the inboard trailing edge represented the highest hingemoment and highest power application, and the thrust reverser represented the hottest operating environment in the flight control system. It was felt that, if satisfactory actuation devices could be provided for these applications, a satisfactory actuation device could be provided and defined for all flight control and utility functions on the aircraft without the need for detailed design in every instance.

2.3.1.7 Baseline Utility Actuator Requirements - In the course of the analysis leading to the preparation of Table 9 certain changes were made which impacted the basic study. The changes made were as follows:

1. It was decided that the brake system would remain hydraulic for both versions of the study aircraft (hydraulic and electrical). The reasons for this decision are discussed in paragraph 4.2.1.2.

2. It was decided that the standard (UARRSI) inflight refuel receptacle would be used in both aircraft I and II. The impact of this decision is discussed in paragraph 4.2.1.3.

3. As discussed in paragraph 2.3.1.1 the ground cooling fuel heat sink door actuation was eliminated as a trade study item. Table 2-8 reflects this fact by noting that this function is always electrically operated.

4. As discussed in paragraph 2.3.1.2 the canopy was eliminated as an actuation function.

5. Table 2-8 reflects the revised interface load requirements resulting from the decision relating to utility engine actuation functions discussed in paragraph 2.3.1.3.

TABLE 11. ACTUATOR OUTPUT LOAD REQUIREMENTS

REDUNDANCY REQUIREMENT	PERCENT OF OUTPUT LOAD (ESTAB. IN NA-79-378-12, TABLE 3-3) REQ'D PER ACTUATOR	ACTUATION FUNCTIONS AFFECTED	ACTUATORS PER SURFACE
FAIL OPERATE-FAIL OPERATE-FAIL SAFE (FO^2-FS)	70%	(1) INBOARD FLAPS (1) MIDSPAN FLAPS	3 3
FAIL OPERATE-FAIL OPERATE-FAIL SAFE (FO^2-FS)	50%	(1) UPPER RUDDER (1) LOWER RUDDER & SPEED BRAKE	3 3
FAIL OPERATE-FAIL SAFE (FO-FS)	100%	(2) CANARD (1) AILERON	2 2
FAIL OPERATE-FAIL SAFE (FO-FS)	(4) $33\frac{1}{3}$%	(3) LEADING EDGE FLAP	6

- FAIL SAFE DEFINED AS FOLLOWS:
  (1) LOCKED IN TRAIL OR FLOATING WITH DAMPING
  (2) LOCKED IN FAILED POSITION
  (3) LOCKED IN BLOWBACK POSITION
- TABLE 3-3 (REF. NA79-378-12) LOADS ARE CLARIFIED AS FOLLOWS:
  (4) LOADS ARE THOSE FOR THE COMPLETE LEADING EDGE FLAP ON ONE WING (POWERED BY 6 ACTUATORS). AS AN ALTERNATE THE FLAPS ON EACH WING MAY BE BROKEN DOWN INTO 3 SEGMENTS POWERED BY 2 ACTUATORS PER SEGMENT

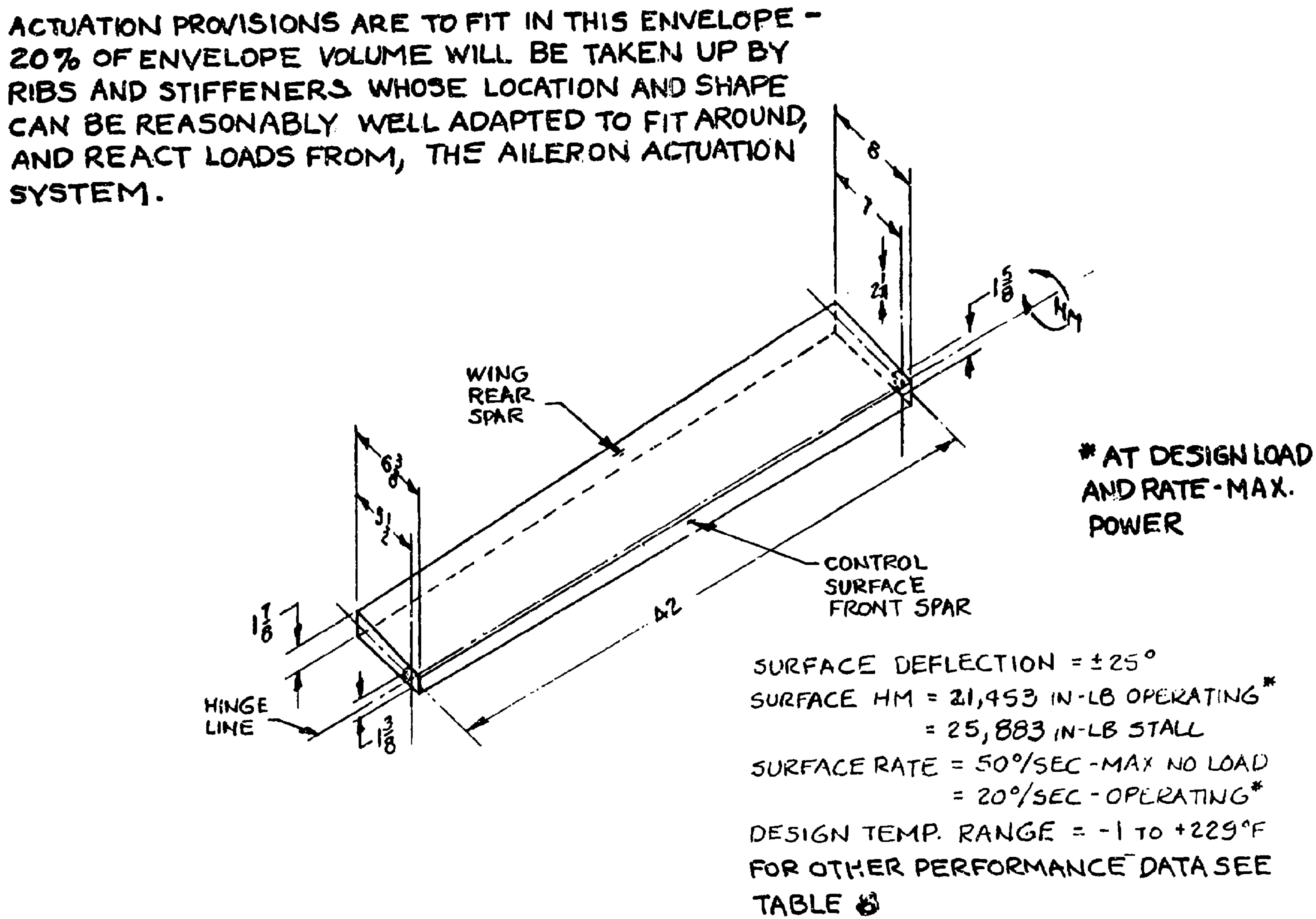
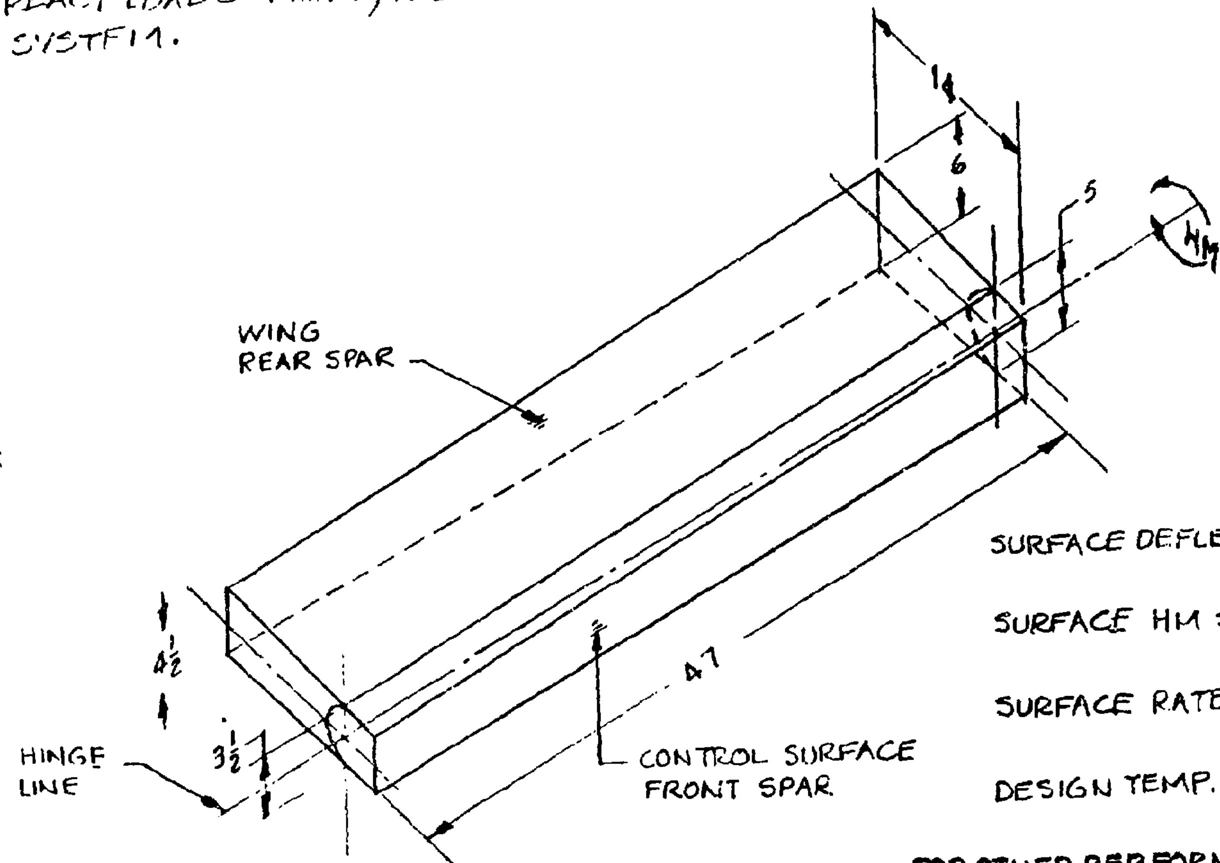


Figure 13. Outboard Trailing Edge (Aileron) Installation Envelope.

ACTUATION PROVISIONS TO FIT IN THIS ENVELOPE - 20% OF ENVELOPE VOLUME WILL BE TAKEN UP BY RIBS AND STIFFENERS WHOSE LOCATION CAN BE REASONABLY WELL ADAPTED TO FIT AROUND, AND REACT LOADS FROM, THE SURFACE ACTUATION SYSTEM.



* AT DESIGN LOAD AND RATE - MAX. POWER

SURFACE DEFLECTION = -30° (UP)
+45° (DN)

SURFACE HM = 161,664 IN-LB OPERATING*
215,552 IN-LB STALL

SURFACE RATE = 100°/SEC. - MAX. NO LOAD
= 50°/SEC. - OPERATING*

DESIGN TEMP. RANGE = -.1 TO +229°F

FOR OTHER PERFORMANCE DATA SEE TABLE 6

Figure 14. Inboard Trailing Edge Surface (Flap) Instal. Envelope.

6. Table 9 defined the types of actuators which will be used in aircraft I (electrical) and aircraft II (hydraulic). The selection of these actuator types is based on the analysis made in paragraph 4.2.1.6.

2.3.1.8 <u>Baseline Flight Control Actuator Requirements</u> - Table 12 was created as an amplification of Table 8. It added data defining "Power at Interface" (explained in footnotes of Table 12), data on "Actuator Type" and additional functional information in the comments column which did not appear in Table 8.

TABLE 12. BASELINE FLIGHT CONTROL ACTUATOR REQUIREMENTS

FUNCTION	TRAVEL	LOAD CHARACTERISTICS			DESIGN ① POWER/ACT. AT SURFACE	INTERFACE DESIGN ② POWER/ACT.	FAILURE RESPONSE ④	NO. ACT. REQ'D PER AC	ACTUATOR TYPE ⑤		COMMENTS
		STALL ③	OPERATING (DESIGN)	LOAD/RATE (DESIGN)					HYD	ELECT.	
INBOARD FLAP	-30° +45°	151,000 IN-LB	113,000 IN-LB	50°/SEC	11.15 KW (14.94 HP)	14.87 KW (19.93 HP)	FO^2 FS	6	MPH	MPH	POWER TRAIN EFF. = .75
MIDSPAN FLAP	-30° +45°	30,000 IN-LB	23,000 IN-LB	50°/SEC	2.27 KW (3.04 HP)	3.02 KW (4.05 HP)	FO^2 FS	6	MPH	MPH	POWER TRAIN EFF. = .75
AILERON	±25°	26,000 IN-LB	21,000 IN-LB	20°/SEC	0.83 KW (1.11 HP)	1.11 KW (1.49 HP)	FO FS	4	MPH	MPH	POWER TRAIN EFF. = .75
UPPER RUDDER	±20°	9,000 IN-LB	7,000 IN-LB	30°/SEC	0.41 KW (0.56 HP)	0.55 KW (0.74 HP)	FO^2 FS	6	MPH	MPH	POWER TRAI.. EFF. = .75
LOWER RUDDER	±20°	13,000 IN-LB	10,000 IN-LB	20°/SEC	0.39 KW (0.53 HP)	0.52 KW (0.70 HP)	FO^2 FS	6	MPH	MPH	POWER TRAIN EFF. = .75
LEADING EDGE FLAP	-0 +20°	173,000 IN-LB	153,000 IN-LB	2.4°/SEC	0.75 KW 1.00 HP)	1.77 KW 2.37 HP)	FO FS	12	MPH	MPH	POWER TRAIN EFF. = .42
CANARD	15.36 IN.	2,400 LB	2000 LB	2.3 IPS	0.17 KW (0.23 HP)	0.23 KW (0.31 HP)	FO FS	4	HLA	MBS	POWER TRAIN EFF. = .75 FOR (MBS), FOR (HLA) SEE ③
THRUST VECTOR VANE	±2.10 IN.	75,314 LB	64,017 LB	0.31 IPS	0.72 KW (0.97 HP)	0.96 KW (1.28 HP)	FO FS	2	HLA	MBS	POWER TRAIN EFF. = .75 FOR (MBS), FOR (HLA) SEE ③

① POWER PER ACTUATOR DELIVERED AT THE SURFACE OR FUNCTION BEING OPERATED

② POWER DELIVERED AT A MOTOR TO GEAR TRAIN INTERFACE WHERE THE GEAR TRAIN CONSISTS OF A GEARHEAD, A SHAFT AND A MECHANICAL HINGE FOR (MPH) AND OF A GEAR TRAIN AND A BALLSCREW FOR (MBS)

③ USE STALL LOAD AND PRESSURE TO DETERMINE PISTON AREA AND USE RATE TIMES AREA TO DETERMINE POWER FOR (HLA)

④ FO^2 = FAIL OPERATE - FAIL OPERATE
FO = FAIL OPERATE
FS = FAIL SAFE

⑤ HLA = HYDRAULIC LINEAR ACTUATOR
MPH = MOTOR - POWER HINGE
MBS = MOTOR - BALL SCREW

2.4 Trade Study Ground Rules

2.4.1 Hydraulic System Burst Factors - Burst factor is essentially a safety factor applied to hydraulic system components and represents the ratio between the system's normal operating pressure and the minimum pressure at which any component in the system will burst. The higher the factor the thicker the walls of pressure containing components, such as valves, tubing, fittings etc., tend to become. For this reason burst factors are a major determinant of hydraulic system component size and weight. As discussed later (paragraph 4.2) 8000 PSI was selected as the normal operating pressure for the hydraulic system in aircraft II. Since 8000 PSI is considerably higher than the current standard operating pressure of 3000 PSI, it was felt that a study should be conducted to determine whether the burst factor for an 8000 PSI system should be higher, lower, or the same as the burst factor of 4.0 currently used for the design and test of 3000 PSI systems. Such a study was conducted.

The results of this study are summarized in Table 13 and show that a burst factor slightly less than 3 can be used in an 8000 PSI system (a burst factor of 4 is the standard requirement for conventional 3000 PSI hydraulic systems). The study was based on an aircraft pressure change duty cycle as follows:

1. 5000 system start-up, shut-down cycles

2. 200,000 rapid valve closures and openings

3. 1 X 10^{11} pump ripple cycles

Item 1 above represented the number of times the system was pressurized and depressurized, both prior to and after flight and on the ground during ground servicing. Item 2 above represented a composite of all the wide ranging pressure excursions which occur in the system predominantly as a result of rapid opening and closing valves. The composite pressure changes (200,000 cycles) were based upon rapid valve closure in a line in which fluid was flowing at 25 ft per sec. It is interesting to note that the magnitude of the pressure perturbations caused by valve action (whether large or small) were primarily a function of fluid flow velocity and were relatively little effected by the increased bulk modulus and density which resulted from increasing the system's operating pressure from 3000 to 8000 PSI. Because flow velocity had no direct relationship to rated system operating pressure, it was assumed that flow velocities were equal (i.e. 25 ft/sec) for all system pressure levels considered in this study. As a result the pressure variations (and hence the stress cycling which occurs in the tubing) were of nearly constant amplitude independent of the system's rated pressure.

As an example; the valve operation pressure variation cycle amplitude (See Table 13 for an 8000 PSI system was only 12% greater than that for a 3000 PSI system. The net effect of this was that the valve operation pressure variations degraded the tubing much less (in terms of fatigue life) in the relatively heavy walled 8000 PSI tubing than they did in the 3000 PSI tubing.

Item 3 above represented a composite of the pump ripple induced pressure variations plus those which might have been induced by other sources such as hydraulic motors and servo valves. These had the same general characteristics as valve operation in that the magnitude of this type of pressure change did not increase in proportion to system pressure increase. As an example, in Table 13 when the rated pressure was increased from 3000 PSI to 8000 PSI (a 2.66 factor), the pump ripple amplitude, as measured in actual pump tests (Reference

TABLE 13. BURST PRESSURE COMPARISON

Duty cycle based on ATS aircraft, Ref NA TFD-81-110

Pressure (PSI)	Burst Factor	Condition	Pressure Range (PSI)	% U.T.S.		"R" (1) Factor	Cycle (2) Life $K_t = 3$	Actual Duty Cycles	% Life Used	% Life Used Total	Plot Point No
				Min	Max						
3000	4.0 (12000) Psi	Start/Shutdown	0-3000	0	25.0	0	$5x10^5$	$5x10^3$	1.0		1
		Valve Operations	1864-4163	15.5	34.5	+0.45	$4.5x10^5$	$2x10^5$	44.4		2
		Pump Ripple	2825-3175	23.5	26.5	+0.89	∞	$1x10^{11}$	0	45.4	3
8000	3.0 (24000) Psi	Start/Shutdown	0-8000	0	33.0	0	$9x10^4$	$5x10^3$	5.6		4
		Valve Operations	6712-9288	28.0	38.7	+0.72	$10x10^6$	$2x10^5$	2.0		5
		Pump Ripple	7750-8250	32.3	34.4	+0.94	∞	$1x10^{11}$	0	7.6	6

1 "R" Factor, $R = \frac{\text{Minimum Stress}}{\text{Maximum Stress}}$

2 K_t = 3 (Notch Factor) Determines which Goodman diagram is used. Conservatively based on R.I. experience

3), increased from 350 PSI to only 500 PSI (a 1.43 factor). This also tended to reduce the fatigue life impact of this type of pressure variation on the tubing as rated system pressure increased.

It can be seen in Table 2-13 that, working with the duty cycle outlined above, the percent of tubing life used was 45.4% at a 4 burst factor in 3000 PSI system. When the system pressure was increased to 8000 PSI and the burst factor was simultaneously reduced to 3 the life used reduced to 7.6%. Actually the burst factor could be reduced to 2.5 before the tubing performance would be reduced to equal that in the 3000 PSI system. However, to err on the conservative side, and to allow for increases in flow velocity in some 8000 PSI subsystems, a burst factor of three was used in the study.

The data for the percent life used column in Table 13 was derived from the ratio of the cycles imposed on the tubing (ninth column in Table 13) to the life expectancy of the tubing under the type of cycles imposed (eighth column in Table 13). The actual life expectancy data was derived from the appropriate Goodman diagram (Figure 15), with the data point number shown on the Goodman diagram being the same as the "Plot Point Number" shown in the last column of Table 13. The Goodman diagram was based on 3AL 2.5V titanium with an ultimate tensile strength (F_{tu}) of 130,000 PSI and an assumed notch factor (K_t) of 3. This notch factor was representative of the stress risers which occur in tube to fitting joints and around oval crossection bends. These were the points where experience has shown that most tube failures occur.

2.4.2 Hydraulic System Design Pressures - Based largely on the results of the burst factor study a revised set of design pressure requirements, applicable to an 8000 PSI system, were created. These requirements are shown in Table 14 and are arranged in the same general format as that used in MIL-H-5540. Conventional 3000 PSI system pressures, from MIL-H-5540, are shown in the table for comparison purposes. It will be noted that the percent system pressure for an 8000 PSI system was in no cases greater, and in several cases less, than those used in a 3000 PSI system. This reduction in pressure ratio, where it occurs, was justified based on an extension of the burst factor data generated in paragraph 2.3.1.

2.4.3 Hydraulic System Design Criteria - The definition of the baseline hydraulic system for aircraft II was based on the following criteria:

1. System configuration was to be in accord with MIL-H-5440.

2. MIL-H-83282 hydraulic fluid was to be used where fluid soak temperatures would not be less than -20°F, MIL-H-5606 was to be used at lower temperatures.

3. Rated system pressure was to be 8000 PSI

4. Aircraft mission was to be as defined in paragraphs 2.1.1, 2.1.2, and 2.1.3.

5. Aircraft functional configuration was to be in accord with Figure 6 (Sheet 1 and 2) except as revised in this report.

6. Aircraft physical configuration (inboard profile) was to be as shown in Figure 22 except as modified elsewhere in this report.

7. Actuation loads were to be as defined in Table 8 and 9 and as expanded upon in Table 12.

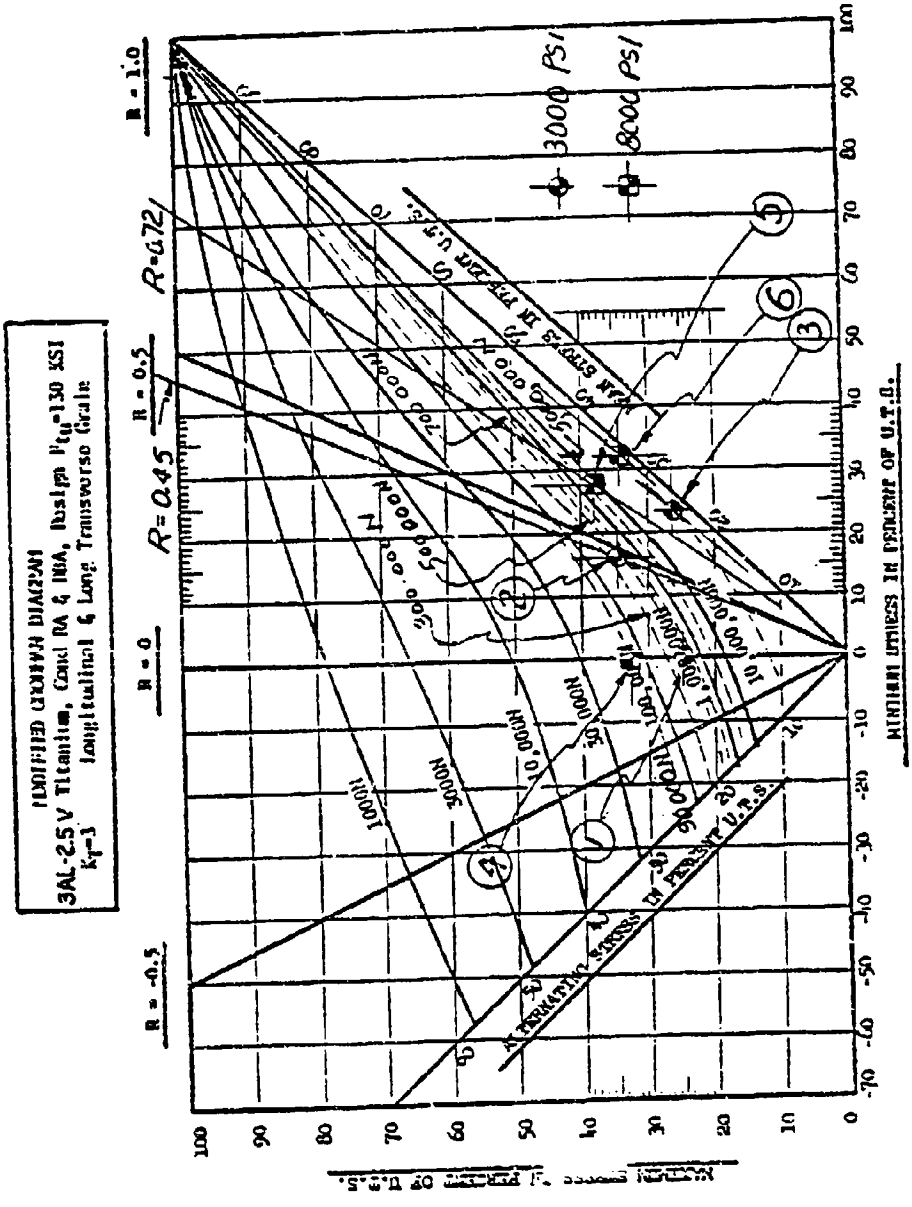


Figure 15. Modified Goodman Diagram.

TABLE 14. SYSTEM PRESSURES

1 of 2

CHARACTERISTICS	3000 PSI SYSTEM		8000 PSI SYSTEM	
	NOMINAL PRESSURE	% SYST. PRESSURE	NOMINAL PRESSURE	% SYST. PRESSURE
PUMP (VARIABLE VOLUME)				
a. PUMP UNLOADING PRESSURE	3000 PSI	- - -	8000 PSI	- - -
b. MAX. LIMIT OF FULL FLOW PRESSURE	2950 PSI	- - -	7900 PSI	- - -
c. MAX. SYSTEM RELIEF VALVE SETTING AT MAX. SYSTEM FLOW	3850 PSI	- - -	8850 PSI	- - -
THERMAL RELIEF VALVE SETTING (MAX.)				
a. EQUAL TO SYSTEM RELIEF VALVE SETTING PLUS VALUES NOTED	150 PSI	- - -	150 PSI	- - -
PROOF PRESSURE (MIN.)				
a. LINES FITTINGS AND HOSES	6000 PSI	200 %	14000 PSI	175 %
b. COMPONENTS CONTAINING GAS UNDER PRESSURE	6000 PSI	200 %	14000 PSI	175 %
c. PUMP SUCTION AND CASE DRAIN LINES COMPONENTS AND RESERVOIRS (BOOT STRAP TYPE)	150 % OF RESERVOIR OPERATING PRESSURE	- - -	150 % OF RESERVOIR OPERATING PRESSURE	- - -
d. COMPONENTS UNDER SYST. PRESSURE ONLY AND PRESS. CIRCUITS (INCLUDING LINES, FITTINGS, AND HOSES WHICH ARE A PART OF THE COMPONENT)	4500 PSI	150 %	12000 PSI	150 %

TABLE 14. SYSTEM PRESSURES (CONCL) 2 of 2

CHARACTERISTICS	3000 PSI SYSTEM		8000 PSI SYSTEM	
	NOMINAL PRESSURE	% SYST. PRESSURE	NOMINAL PRESSURE	% SYST. PRESSURE
PROOF PRESSURE (MIN.) (CONTINUED)				
e. COMPONENTS UNDER RETURN PRESS. ONLY AND RETURN CIRCUITS (INCLUDING LINES, FITTINGS, AND HOSES WHICH ARE A PART OF THE COMPONENT)	2250 PSI	75 %	6000 PSI	75 %
BURST PRESSURE (MIN.)				
a. LINES FITTINGS AND HOSES	12000 PSI	400 %	24,000 PSI	300 %
b. COMPONENTS CONTAINING AIR AND FLUID UNDER PRESSURE	12000 PSI	400 %	24,000 PSI	300 %
c. PUMP SUCTION AND CASE DRAIN LINE COMPONENTS AND RESERVOIR (BOOT STRAP TYPE)	300 % OF RESERVOIR OPERATING PRESSURE	- - -	300 % OF RESERVOIR OPERATING PRESSURE	
d. COMPONENTS UNDER SYST. PRESSURE ONLY AND PRESS. CIRCUITS (INCLUDING LINES, FITTINGS, AND HOSES WHICH ARE A PART OF THE COMPONENT).	7500 PSI	250 %	20,000 PSI	250 %
e. COMPONENTS UNDER RETURN PRESS. ONLY AND RETURN CIRCUITS (INCLUDING LINES, FITTINGS, AND HOSES WHICH ARE A PART OF THE COMPONENT)	4500 PSI	150 %	12,000 PSI	150 %

8. Tubing material for all pressure return and suction tubing was to be 3AL 2.5V titanium.

9. The landing gear system was to meet operating requirements at -40°F and all other systems were to meet operating requirements at+20°F.

2.4.4 Peak Actuation Loads - It was assumed, based on experience, that during maximum actuation power demand for any mission segment no more than 2/3 of all actuators would "peak" simultaneously. It was, therefore, assumed that, for both aircraft I and II, the peak actuation power demand would be 2/3 of the theoretical sum of all short term actuation power demands occurring during a given mission segment.

2.4.5 Actuation Configuration - Where rotary shaft input power using motors was indicated as the best solution for aircraft II actuation functions, motors were also to be used on aircraft I. All elements downstream of the common mounting pad (interface) for these motors were to be identical between aircraft I and aircraft II. All actuation functions which use hydraulic linear actuators in aircraft II were to use linear ballscrew actuators attaching to the same end points in aircraft I.

2.4.6 Trade Study Aircraft - The trade study aircraft configurations were designated as aircraft I and aircraft II. Aircraft I was the "all electric" power-by-wire version in which essentially all secondary power on board the aircraft was generated and utilized electromechanically and was transmitted electrically. Aircraft II was the baseline aircraft which employed a conventional power split between hydraulic and electrical power generation, distribution and utilization. Figure 16 flowcharts the candidate configuration concepts. The basic concepts, which were traded, are represented by boxes 2 and 6 in Figure 16 with box 6 representing the baseline configuration. The other concepts shown in figure 16 (boxes 1, 3, 4, and 5) were examined in some depth but were gradually eliminated as the study progressed.

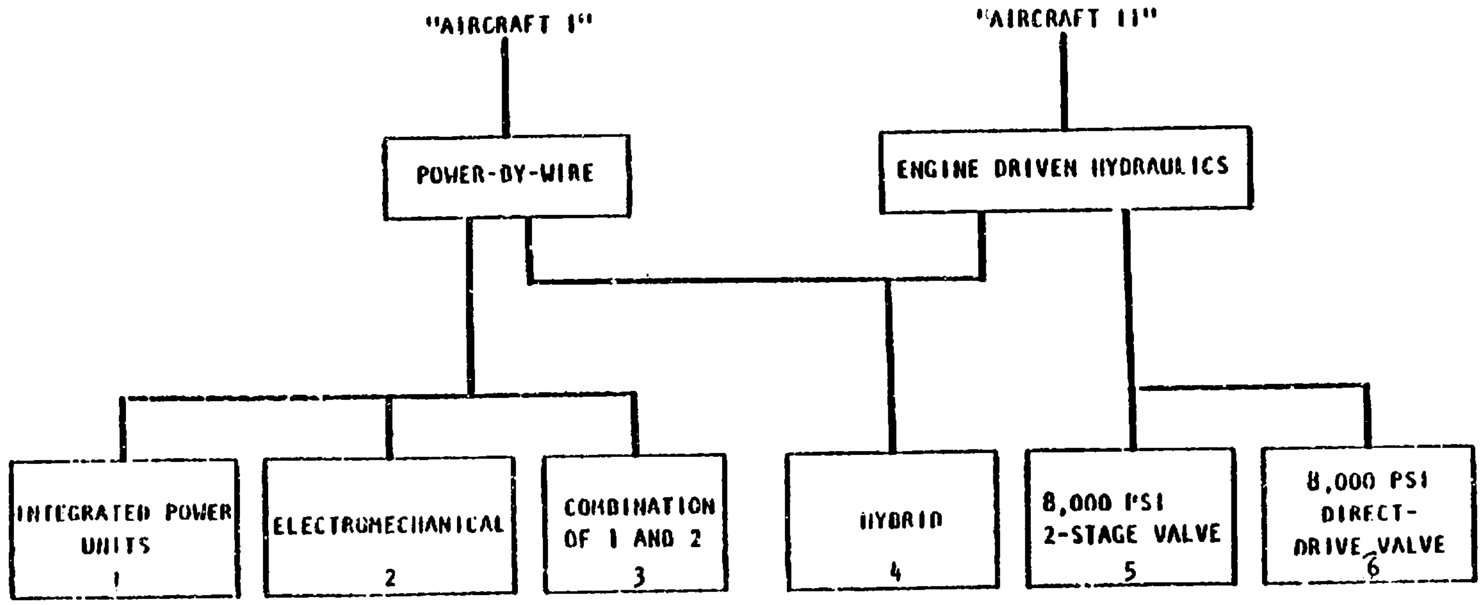


Figure 16. Candidate Configuration Concepts.

3.0 CONCLUSIONS AND RECOMMENDATIONS

3.1 Conclusions - Based on the study results it was concluded that the "All Electric" approach, to aircraft secondary power system design, did not provide a viable alternative to the more conventional Hydraulic-Electric when applied to aircraft of the 1990 + time period. The data presented in paragraphs 4.0 and 5.0 showed quite clearly that the "All Electric" approach was deficient in all major catagories (i.e., weight, maintainability, reliability and life cycle costs) relative to the advanced (8000 psi) version of the conventional Hydraulic-Electric approach.

During the conduct of the study, Rockwell was aware that the study results were probably not going to favor the application of the "All Electric" approach to the ATS aircraft. However it was felt by Rockwell personnel that, even though not satisfactory for a compact high performance aircraft such as the ATS, the all-electric approach might prove highly desireable for a large subsonic aircraft. This optimism stemmed from three factors as follows:

1. The belief that the Maintainability and Reliability advantages inherent to the single power type distribution system (Electrical) versus the dual power type distribution system (Hydraulic-Electrical) would more than offset some of the known Reliability deficiencies characteristic of certain power handling electrical components (switches and power control inverters).

2. The belief that the known weight penalties associated with the need to use power control inverters in the selected (270 VDC) electrical system could be more than offset, as the aircraft grew larger, through the supposed weight savings associated with transmitting power longer distances through wire as opposed to transmitting the same, or slightly greater, power through both hydraulic tubing and parallel wire routings.

3. Given an assumed Maintainability/Reliability improvement and a weight reduction as the aircraft became larger, it was believed that, at some point (particularly if the loads, and hence inverter weights, did not increase proportionately because the aircraft was no longer supersonic) the life cycle costs would cross over and favor the "All Electric" approach.

Unfortunately these optimistic presumptions proved to be incorrect. The basis for the foregoing statement was provided by a brief study conducted by Rockwell to roughly evaluate the presumed advantages of the large subsonic aircraft. This study assumed an aircraft having a gross takeoff weight (GTOW) more than ten times the GTOW of the ATS (i.e., 400,000 lb GTOW) and gross dimensions (i.e., length, height and wing span) four times those of the ATS. It was further assumed that, because the aircraft was subsonic (0.85M), the power generation (pumping system) and utilization systems (actuators) were essentially identical in power requirements to those of the ATS. The assumed aircraft was very similar to the Lockheed L1011 in all basic characteristics except that the power required for the L1011 was significantly less than that required for the ATS (387 hp for the L1011 versus 461 hp for the ATS). It was felt, however, that the higher power requirement was probably quite representative of an advanced 1990 + control configured large subsonic aircraft. Using these assumptions the only significant difference between the ATS study aircraft (Aircraft II) and the

large subsonic aircraft was the power distribution system. This system was assumed to carry the same power at the same transmission efficiency (pressure drop or voltage drop) from the source to the utilizing function through an average transmission distance which, for the large subsonic aircraft, was 4 times that of the ATS.

The weight of hydraulic plumbing (filled with fluid and transmitting power at equal efficiency) tends to increase by a factor of two when the transmission distance increases by a factor of four. This was derived from the pressure drop curves of reference 34. In contrast the weight of electrical wiring tends to increase by a factor of three under the same circumstances. This was derived from wire performance data contained in reference 35.

Using the ATS as a baseline, it can be seen in Table 38 and Paragraph 4.2.2.9 that the total weight of power transmission elements in Aircraft II is as follows:

Tubing, Fittings and Supports	116.8
Reservoir and Supports	44.3
	161.1
Power Wiring	23.0

Extrapolating these weights to the large subsonic aircraft results in the following:

161 lb X 4 (length factor) X 2 (hyd. factor)	=	1288 lb
23 lb X 4 (length factor) X 3 (elect. factor)	=	276 lb
		1554 lb

From Table 4-12 it can be seen that the total weight of power transmission is 120.3 lb for Aircraft I. Extrapolating this weight to a large subsonic aircraft results in the following:

120.3 lb X 4 (length factor) X 3 (elect. factor) = 1440 lb

From the above it can be seen that the weight saving in power transmission elements, through the use of the All Electric approach, was as follows:

For the large subsonic aircraft	114.0 lb
For the ATS (Aircraft I versus II)	63.8 lb
Δ weight saving-large aircraft versus small aircraft	50.2 lb

Although there was an increased (Δ) weight saving, in going from the small to the large aircraft, the amount of weight saved was too small, by an order of magnitude, to effectively offset the adverse weight impact of the inverters. A weight saving of at least 500 lb would have been necessary to negate the effect of the inverters. From the evaluation of the large subsonic aircraft it was concluded that no significant weight

advantages could be expected from increased aircraft size. Essentially the Weight, Reliability, Maintainability, and Life Cycle Cost penalties associated with the "All Electric" approach would tend to remain the same whether the aircraft was large or small, or high performance or low performance.

3.2 Recommendations - Although this trade study does not indicate that the "All Electric" approach will be viable through the mid 1990s, it is still a very intriguing concept. For this reason it is recommended that work be continued on inverter development. If inverter weight could be halved while reliability was improved the "All Electric" approach would become viable in a wide range of applications.

The trade study indicates that power hinge type devices will be increasingly needed as a basic element in primary flight control surface actuation whether the actuation function is powered hydraulically or electrically. However, even though the power hinge seems a perfectly feasible device for this type of application, there is very little background based on actual operating experience, particularly, on a high output flight control duty cycle using long multi-slice small diameter power hinges. It is therefore recommended that multi-slice (at least 15 slices) small diameter power hinges (less than 1.5 in. dia) be developed and tested for thin wing trailing edge control surface applications. Additional characteristics which should be demonstrated during the course of development are as follows:

Frequency Response	20 hz
Stall Hinge Moment	25,000 in-lb
Operating Hinge Moment	20,000 in-lb
Operating (Design Load) Rate	50°/sec
Maximum (No Load) Rate	100°/sec
Minimum Dynamic Stiffness	5×10^5 in-lb/rad
Minimum Hinge Stack L/D	15
Operational Service Life	8000 hrs

Demonstrated ability to function in the presence of wing (hinge line) flexing.

Because the study indicates that the 8000 psi hydraulic system approach offers the greatest potential for low Life Cycle Cost power systems in the mid 1990 time period, it is recommended that development be pushed in at least two critical areas. These areas are as follows:

1. Small Motors - Small high frequency response 8000 psi servo-motors should be demonstrated. These servo-motors should have the following general characteristics:

Speed-max.	20,000 rpm
Torque-stall	21 in-lb
Frequency Response	Suitable for operating with power hinge described above
Envelope	2.5" wide 1.5" high 3.5" long

2. Dynamic Seals - Although adequate data is available on short term seal life (up to 500 hr), more long term testing (up to 8000 hr) is needed to assure that 8000 psi systems will meet the projected reliability goals.

4.0 TRADE STUDY AIRCRAFT DESCRIPTION

4.1 Aircraft I - Aircraft I was the "All Electric" version of the baseline study aircraft. In this version all the power required for secondary functions was extracted from the primary power sources (engine or APU), in the form of rotary shaft power and was either utilized directly (i.e. as in the case of the ECS compressor mounted on the airframe mounted accessory gearbox) or was distributed to the various utilizing functions in the form of electrical power. There was no hydraulic or pneumatic power generation and distribution system on board the aircraft except for two integrated actuator package (IAP) type systems represented by the brakes and the inflight refuel receptacle.

4.1.1 Power Generation and Starting - Figure 4-1 shows the arrangement of the power generation and starting system for aircraft I. It shows that power for all secondary function was extracted from the engine through the PTO shaft with the exception of a very small amount of power (4.27 HP Max) which was withdrawn from the engine in the form of bleed air to maintain pressure in the cockpit and avionics compartments. Based on the data given in paragraph 2.2.6 the required power for starting was 120 HP (89.5 KW). The generator shown in Figure 17, having a 60 KW rating both as a generator and a starter, would meet this requirement because, operating within its normal 150% (90 KW) overload capability for 2 minutes, it would easily meet the 120 HP (89.5 KW) for 35 seconds starting requirement. (See further discussion in paragraph 4.1.2.2) Numerous power generation/starting approaches were examined before the approach shown in Figure 17 was selected as the baseline. The major potential approaches examined and compared were as follows:

1. Conventional 400 Hz AC power (115/200 V)
2. Double voltage 400 Hz AC power (230/400 V)
3. Integrated drive starter generator type of constant speed drives
4. 270 Volt DC power
5. Mixed 270 VDC and conventional 400 Hz AC power
6. 270 Volt DC starting with drain and fill torque converter
7. Generators integrated with and buried in the engines

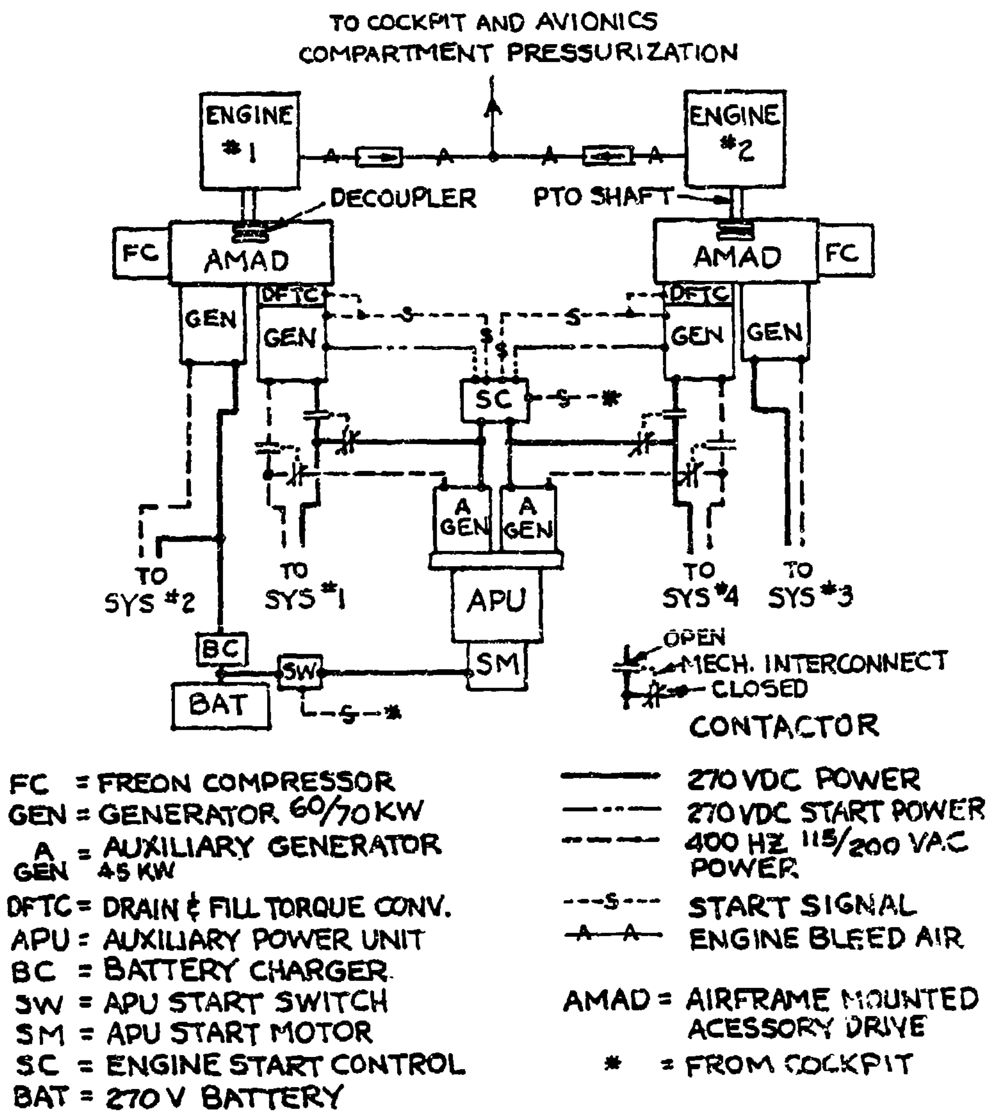


Figure 17. Aircraft I Secondary Power Generation System.

4.1.1.1 Conventional vs Double Voltage 400 HZ ZC Power Comparison - A double voltage system was compared to a standard voltage system as a potential power generation system approach for use in the study aircraft (see items 1 and 2 above). The double voltage system was defined as a 230/400 volt, 400 Hz, 3-phase AC system. The double voltage approach was a recent development and its single major practical application has been on the B-1 Bomber aircraft. The standard voltage system was a 115/200 volt, 400 Hz, 3-phase, AC system. Standard voltage systems have been in service on aircraft for a number of years and were accepted as reliable and safe.

Subjects of concern considered during the comparison study were safety (personnel and aircraft), corona, electromagnetic interference, utilization equipment and electrical components (E.G., relays, connectors, wire, etc.) and weight.

4.1.1.1.1 Safety - When evaluating personnel safety, a reasonably safe "let go" voltage for man, assuming wet contacts, was considered to be between 10 and 21 volts. Since the potentials of the standard voltage system and the double voltage system were both considerably above the "let go" voltage, the hazards were not considered to be significantly different for either voltage.

With regard to aircraft safety, there was no reason to believe that a fault or short circuit should occur with any greater frequency on a double voltage system than on a standard voltage system. This assumed that the design incorporated terminal spacing, insulation characteristics, etc., commensurate with double voltage system requirements. As to the effect of a fault, the double voltage generator would limit the steady state fault current (beyond ½ cycle) to roughly half of that in a standard voltage system. The amount of power to the fault would be approximately equal to that in a standard voltage system, and therefore, the consequences, or result, of the fault should be approximately equal regardless of the system voltage used. In summary, from a safety standpoint, the precautions and procedures necessary for a double voltage system were not considered significantly different from those required with a standard voltage system. The hazards present in either system could be minimized by good engineering design and observance of good safety practices.

4.1.1.1.2 Corona - Corona would occur whenever the voltage gradients between electrically energized electrodes exceed a critical value. Its onset voltage would be a function of ambient pressure, temperature, insulation material and thickness, and time. Undesireable effects of corona would be insulation degradation, interference, and power loss. The altitudes where corona may become a problem would be above 50,000 feet, with the most critical altitudes being between 100,000 and 200,000 feet. Generally speaking, voltages of 300 volts (peak) would not cause corona onset even at altitudes of 100,000 feet. The maximum transient voltage on a double voltage system is 508 volts (peak) and would only exist during abnormal operation of the generating system. In summary, the potential for presence of corona existed with the double voltage system, however, the proper conditions occured only under random and rare circumstances.

4.1.1.1.3 EMI - The primary source of interference from the electrical power and distribution system was through electrostatic or a electomagnetic coupling to a susceptible signal circuit. Power input to a black box was assumed to be adequately filtered and not a factor. The magnitude of the electromagnetic coupling would be determined by the change in current. Since the circuit currents in a double voltage system would be approximately half of that in a standard voltage system, it was anticipated that there would be a decrease in electromagnetic coupling.

The effectiveness of electrostatic coupling would be largely determined by the voltage. Therefore, electrostatic coupling might be greater in the double voltage system. This type of coupling could usually be adequately controlled by proper use of shields .

In summary, it was felt that the total electromagnetic interference due to a double voltage system should be no greater than that of a standard voltage system.

4.1.1.1.4 Utilization Equipment and Electrical Components - New utilization equipment could be designed to operate directly from a double voltage input without any increase in weight or volume. There were felt to be no significant technical barriers to the development of this equipment.

Electrical components rated for operation at 230 volts presented no significant problems. Some equipment (switches, connectors, wire, etc.) could be uprated for double voltage operation without any change or penalties. Other equipments would require minor modifications.

4.1.1.1.5 Weight - During its design, numerous configurations of the electrical power generation and distribution system were studied to determine the system with the maximum weight advantage for the B-1 aircraft. These studies (Reference 15) showed that for a large aircraft (395,000 Lbs. take-off weight) with a large electrical system (345 KVA), reasonably large weight savings could be realized with a double voltage electrical system. Most of the weight saving was derived from the use of long runs of primary power feeder cables and resulted from less copper being required to carry the reduced current characteristic of the double voltage system.

The weight advantage which could be realized by utilizing a double voltage system, instead of a standard voltage system, on aircraft I was relatively small and could be determined by a detailed weight analysis of the major area bus feeders, transformers, and secondary power rectifiers. For an aircraft the size of aircraft I (36,043 Lbs. take-off weight), with a "worst-case" AC load of approximately 50 KVA the weight savings would not have been substantial enough to have been cost effective. In order to have been cost effective, all of the double voltage utilization equipment would have to have been available in 1990 for use in aircraft I as off-the-shelf equipment or with only minor modification. Since there was no discernable trend to the development of double voltage utilization equipment at the time of this study, a standard voltage electrical system was considered most acceptable.

4.1.1.1.6 Summary - The effects of certain technical considerations (i.e., safety, corona, EMI, utilization equipment and electrical components) did not legislate against the use of a double voltage electrical system. However, for air vehicles the size of aircraft I, the weight saving potential available, through the use of double voltage, did not appear sufficient to justify the cost of developing "new" utilization equipment. Therefore, further consideration of double voltage systems was dropped.

4.1.1.2 Generators Integrated with Engines - The possibility of integrating the generators with the engine was looked at in considerable depth. The engine used in the ATS aircraft is a 1995's type twin spool, low bypass ratio, variable cycle engine. Based on this design the apparent practical locations for the generators were considered to be confined either to installation in the engine inlet bullet nose or mounted as an external accessory on the engines' waist. A third possibility, which was integration of the generator with N_1 or N_2 spool shaft, did not appear practical primarily due to the extremely poor accessibility of such an installation.

The engine bullet nose location had been used in the past and had proved reasonably feasible for a single generator of small size. However, considering the fact that this engine installation required the mounting of two generators, and both of large size, the engine bullet nose approach appeared much less desireable. The engine bullet nose on the ATS engine had a useable inside diameter of 6.25 in. which was much too small to seriously consider side by side mounting of the generators. Even if the generators were mounted in tandem, their diameter (10 in.) would have increased the bullet diameter and, hence, the engine diameter thus offsetting one of the primary virtues of the generator's integration with the engine; i.e. reduced installed engine/inlet frontal area. The extreme length of a tandem installation would have futher amplified an already existing problem characteristic of previous bullet nose installations which was excessive cantilever vibration. As final factors legislating against selection of this approach, bullet nose installations had the following additional disadvantages:

1. Poor accessibility with engine installed

2. A significant increase in the possibility of engine FOD damage

3. Poor acess, as a starter generator, to the high speed (N_2) spool which is the spool which must be powered during starting.

An engine waist installation was considered as being essentially identical to an AMAD installation. The selection between the two would go to that approach which provided the best accessibility and the smallest increase in engine/inlet system frontal/wetted area.

Because the engine used in this application was a low bypass ratio type it tended to be barrel shaped and have very little "waist". For this reason any acessories mounted on the engine even in the waist area, tended to increase the engine's frontal area by an amount at least equivalent to the frontal area of the item being mounted. It can be seen however, when looking at the ATS inboard profile (Figure 22) that, due to the nature of the inlet duct (above and outboard on the fuselage) and its relation to the wing (see station 560 crossection), a forward extending PTO and AMAD setup would allow the installation of two large generators, in place of the items shown as 4 and 8 on the inboard profile, with no significant increase in engine/inlet system frontal area or wetted area. As pointed out else where in this report, and as shown in Figure 17, the AMAD also drives the freon compressor. The ability to drive the freon compressor by direct mechanical drive, rather than by an intervening electrical link, was felt so important efficiency-wise that, even if the generators were mounted on the engines' waist, an AMAD would have had to have been provided for driving the compressor by itself or the three units (two generators and one compressor) would have had to have been mounted as a group on the engine's waist. This latter approach represented a prohibitive penalty because, in addition to the three components's large size, there was the fact that they had to be intermixed with other engine ancillary equipment, such as fuel controls, with the result that there was a large adverse impact on, not only the engine/inlet system frontal and wetted area, but also on accessibility and maintainability. Based on the foregoing considerations, the AMAD approach was selected as offering the best access and maintainability in combination with a minimum increase in frontal and wetted area.

4.1.1.3 270 Volt DC Power Versus Mixed 270 VDC/400 HZ AC Power - The primary source of power on almost all civil and military aircraft for the past 25 years has been based upon the distribution of 115/200 volts, 400 Hz AC power. With the passage of time, this system has been widely accepted and has shown a steady increase in reliability and specific power output. As will be seen later (paragraph 4.1.5.1), the primary control surfaces (in the study aircraft-aircraft I) are activated by power hinge actuators utilizing 270 volt DC samarium-cobalt, brushless (permanent magnet) drive motors. Due to the importance of this system to aircraft I operation and because the actuation loads are a major portion of the aircraft's total continuous load (approximately 66 percent during combat), it was decided that a 270 volt DC power generation and distribution system would provide the most efficient power source for this type of equipment.

Numerous analyses have been performed relative to the impact on aircraft weight, reliability, safety, and cost if a 270 VDC system were substituted 100% for the conventional 400 Hz AC system (Reference 9, 12, and 13). The conclusions arrived at, from the various studies, were that equipment in the form of brushless DC motors, power semiconductors, solid state switchgear, aircraft cables, and inverters were available for a 270 VDC system and that, in large aircraft, a lighter weight and lower maintenance cost system could be expected through the use of 270 VDC power. (Reference 17, 18 and 19). However, the situation with respect to small high performance aircraft (7.33 G-M2.2) was not made clear. In this type of aircraft the weight fraction of the distribution system tends to become small and the weight fraction of the power output devices tends to become large. Therefore, even though a selection of 270 VDC power had been made for the primary control surfaces (modulated) electric motors, the selection of the best power for the unidirectional and/or non modulated electric motors used on the aircraft was open to further examination.

4.1.1.3.1 Non Modulated Motors - Studies done by others were targeted at the use of 270 VDC power for large non modulated motor loads, such as fuel pumps and blowers, that were normally operated by AC induction motors (Reference 13). The two alternatives available were (1) retain the continuously running AC induction motors and drive them with dedicated inverters that convert the 270 VDC to square wave AC power or, (2) use brushless DC motors; i.e., synchronous motors with permanent magnet rotors and armature windings, controlled through solid state circuits (inverter) to provide commutation and run at adjustable speeds. In either event an inverter was required. At the time these (Reference 13) studies were conducted brushless DC motors were considered to have marginal advantages but had only been developed in sizes of approximately one kilowatt. Recent development in brushless DC motors have made their performance better than that of induction motors, with higher efficiencies and better torque/speed characteristics. In addition, several companies have produced prototype DC motors in ratings up to 12 KW.

Estimates made during this study program suggested that the brushless DC motor, with its inverter, would be lighter and smaller than an electronically (inverter) controlled inductions motor of the same rating. There was a third alternative however, and this was the possibility that the overall system would be lighter and more reliable if 400 Hz AC power was supplied for all non-modulated electric motors. As pointed out earlier both of the above alternatives (1 and 2) for operating motors from a 270 volt DC bus require solid state electronics (inverters) for commutation and control. Due to losses developed during transistor switching and conduction, power inverters require cooling and, since the size and weight of the inverter is usually determined by the type of cooling employed, the inverter and its cooling method become an integral part of the motor evaluation.

To form an initial assessment on the size of the inverters required, if 270 VDC power/was to be used for all motor loads, the study for the design of the surface control systems (paragraph 4.1.5.1) was reviewed. In this study, inverter cooling requirements were determined to be in one of two categories:

(1) Inverters rated at 25 amperes (6.75 KW) or smaller may be cooled by natural radiation and convection. A common design for all inverters in this category was sized at 231 cubic inches (11" x 7" x 3") and weighed 10 lbs.

(2) Inverters rated larger than 25 amperes (6.75 KW) required alternate cooling techniques of which evaporative cooling was selected.

Cooling techniques based on natural radiation and convection were preferred, of course, because of their simplicity and low cost.

Continuous motor loads on the study aircraft included two fuel boost pumps (1.98 KW each during cruise operation, and 2.36 KW each with afterburner), ten fuel transfer pumps (0.64 KW each during cruise operation and 0.74 KW each with afterburner), and two blowers (3.2 KW each) and two pumps (0.87 KW each) in the environmental control system. The weight of a common inverter to support operation of these motors from a 270 volt DC bus was estimated to weigh 6.9 lbs $\left(\sqrt{\frac{3.2}{6.75}} \times 10 = 6.9\right)$ by comparing its power rating with the power rating of the surface control actuation system inverter. As a result of the above analysis, a total inverter weight of 110.4 lbs. (6.9 x 16 = 110.4) was concluded to be necessary to operate the motors from the 270 volt DC bus.

In addition to the inverters required for the continous load function enumerated above, inverters would also have been required for all the low power (6.75 KW) utility functions listed in Table 2-1. This would have required at least 10 more inverters involving an additional 69 lbs weight penalty.

If on the other hand 115/220 V 400 Hz AC power could be supplied to these motors the inverters could be eliminated, and simple on-off or extend retract switches substituted, allowing a net weight saving of at least 127 lbs based on an assumed average switch weight of 2 lbs. Although the use of 270 VDC power for servicing these loads would have reduced duplication and the weight of the power feeder cables (see following paragraph 4.1.1.3.3) it was not felt that the saving in an aircraft of the small size of the study aircraft would have been great enough to offset the 127 lbs inverter weight penalty. Therefore, in consideration of this fact, plus the fact that the continued use and availability of 400 Hz power would be convenient for use in avionics black boxes and lighting equipment and would give added flexibility and adaptability to an advanced aircraft of the 1990's a dual power output (270 VDC and 115/200V 400 Hz AC) power generation system was selected for use in the baseline aircraft I.

4.1.1.3.2 Aircraft Cables - It was apparent that the use of a 270 VDC system had the obvious advantage over the conventional AC system of reducing the aircraft's total cable length. This would have been true because the number of power feeder cables required from the generator to the distribution center, as a minimum, decreased from four to two. Earlier estimates on distribution wiring (Reference 20) assumed that high voltage DC power could be supplied by a single cable from each generator with aircraft structure providing a ground return. In comparison with a conventional 3-phase machine, this meant three cables could be replaced by one. With the increased impact of composite structures, it appeared unwise to assume a structural ground return to be practical.

Accordingly, for the purpose of this study, the DC feeder system between each generator and the distribution center was assumed to consist of a positive and a negative cable and the AC feeder system was assumed to consist of 3 positive cables and a ground return cable.

Using the four wire (3 phase AC power) versus 2 wire (DC power) as a basis, a rough study was made of the comparative weights involved in distributing power to the 16 continuous motor loads discussed in the previous paragraph (paragraph 4.1.1.3.1). It was assumed that the motor loads were distributed approximately evenly between the four power systems. It was further assumed that the power distribution center was located three feet from the generators and that the various motors were located as shown on the aircraft inboard profile (Figure 22).

Based on this study the wire weight saving which would have resulted from using a 270 VDC distribution system, rather than 115/200 V 400 Hz AC system, was approximately 4 lbs. Although the total wire weight saving which might have been expected in this aircraft, through using 100% DC power versus using a system approaching 100% AC power, might exceed this value by an order of magniture (i.e. 40 lbs), the rough study indicated quite clearly that, on an aircraft this small, the wire bundle weight saving potential of this approach was small.

In recent years the properties of, and insulation for, wires and cables have improved substantially. At present Kapton and Tefzel are the lightest available materials. It was assumed that these insulation materials were representative of the types which would be used in a 1990 + aircraft and therefore wires insulated with these materials were used for all weight studies for general airframe application. Studies with respect to the effect of a double voltage system on aircraft parameters such as aircraft and personnel safety, corona, EMI and wire have been discussed previously in this report. The conclusion was that precautions and procedures necessary for a double voltage system were not significantly different from those required with a standard voltage system. The hazards present in either system are minimized by good engineering design and observance of good safety practices.

The selection of a 270 volt value for the high voltage DC portion of the system resulted from the ease with which this voltage could be obtained when full wave rectification was applied to a conventional 115/200 volt 400 Hz 3-phase AC generator. As a result there was little doubt that any thin-wall cable acceptable for operation with a double voltage system (such as that used on the B-1 aircraft) would have had adequate margin for operation at 270 VDC.

4.1.1.3.3 Switch Gear - In todays aircraft, electromechanical devices such as circuit breakers, contactors and relays are used for switching and protection in feeder and distribution systems. For an AC powered electrical system this is no problem. During any mechanical switching action, the arc which occurs across the gap tends to extinguish itself at the "current zero" point of the sine wave. With a 270 volt DC system, however, the arc tends to be self-sustaining (at least 3 seconds) and if not detected and extinguished very quickly by some form of forced commutation could result in severe damage.

Earlier studies recognized the danger of arcing and flashover to be very real. They also showed, by laboratory tests, that under certain fault conditions a self-sustaining arc condition might occur that could not be extinguished by the mechanical switchgear then available As a result of this, even today a reluctance exists on the part of designers to accept the 270 VDC electrical system.

Adequate protection against ground faults is dependent upon rapid detection and isolation of the fault condition. Although this cannot be accomplished with the thermal circuit breaker, there was reason for confidence that high rates or rise in current could be detected and transistor switches developed to break a high current fault before it had risen to extreme levels. In addition to opening the circuit, backup safety could be provided by the capability to de-excite or mechanically decouple the generator very rapidly.

Recent developments in high powered, solid state switching devices were considered sufficiently well advanced that, although they had not replaced conventional components in the AC and DC distribution systems of modern aircraft, they were in good position to do so. The use of solid state technologies was felt to be inevitable and crucial in the 270 VDC system to replace functions previously performed by contactors and thermal circuit breakers in the AC system. As always, considerations for cooling the solid state devices were an added problem.

In order for a sustained arc to occur, two faults must coexist; (1) the fault must occur and (2) the protection system (overcurrent sensing) must fail. The improbability of a double fault, along with the confidence that can be placed in the protection system as a result of advances in semiconductor technology, made it reasonable to accept the 270 VDC system in the study aircraft.

For the purpose of this study, solid state power controllers (SSPC's) were utilized to perform AC and DC load switching and protection functions. Each unit provided the function of a circuit breaker and remotely controlled switch (contactor) in a common module. For small load currents the SSPC was considered to be an "all solid state" device. For SSPC's in the 10 to 400 ampere current range a hybrid (solid state switching plus electromechanical contactor) unit was utilized. All units provided for their own heat rejection without the need for addtional cooling.

4.1.2 Power Generation and Distribution - The power generation and distribution system is shown schematically in Figure 18 and the terms used in the schematic are defined in Page 2 of the Figure. The most significant characteristic of this system, and the one which represented the greatest departure from previous electrical system philosophy, was the fact that the total power system consists of four completely independent power channels. There were no bus tie contactors and each generator was dedicated solely to a particular channel. This approach was used to allow the electrical system to satisfy reliability and redundancy requirements (See Tables 4 and 11). Because the electrical system in this aircraft provided actuation power for the flight control and utility systems as well as acting as a source of control signal power for both, it must have a higher order of reliability than that characteristic of electrical systems of the past such as typified by the electrical system of aircraft II. This arose from the fact that, in picking up the actuation functions which have historically been accomplished hydraulically, the electrical system must duplicate or exceed the hydraulic power system's redundancy. Through long and sometimes bitter experience, it has been found that hydraulic systems, in their historic power supplying functions, must have redundant and absolutely isolated systems. It was found that it must not be possible for a failure in one system to propagate into another system because, if it is possible, it will happen.

Conventional electrical systems using bus tie contactors have always been subject to, and have frequently experienced, "cascade" type failures in which a single failure in one generating system has propagated through all systems wiping out all generated power on the aircraft. Such a system approach would not meet the fail operate, fail safe requirements imposed on the flight control system of the study aircraft. This continued to be true even when using stored energy in the form of batteries or APU's as an emergency power source. To meet the failure and reliability requirements of aircraft I it was felt imperative that at least 3 completely independent dedicated power systems be provided. This would match the redundancy of the hydraulic system used in aircraft II.

After more detailed study it was determined that, although 3 independent power systems were adequate, 4 independent systems more closely approached the optimum for the reasons listed below:

1. The use of four systems was the only practical way to balance engine power extraction loads, generator size, and bus loads so that each was simultaneouly reasonably uniform.
2. Four systems fitted well with the 5 channel (4 channel plus model channel) philosophy used in the aircraft I fly-by-wire control system.
3. The use of four dedicated systems required smaller generators and a lighter weight generating system than that of the conventional "bus tie contactor" approach.

With reference to reason #1 above, it will be seen later that system "5 second" and "continuous" loads could be distributed among the 4 channels so that they vary no more than ± 24% from the mean. This could be done while meeting the load and redundancy requirements of Table 11. It was also true that the minimum loads imposed on the PTO shaft by the electrical power system were determined by the generator's size and its fault clearing capability (approximately 250% of continuous load rating). It was also highly desireable that the PTO extraction loads be essentially identical between the two engines. If they were not, the PTO power train in the engine and AMAD must be designed to the highest load seen by either engine. This was necessary to maintain engine interchangeability. Thus, if the loads applied were allowed to get seriously out of balance, the overall PTO system became unnecessarily heavy.

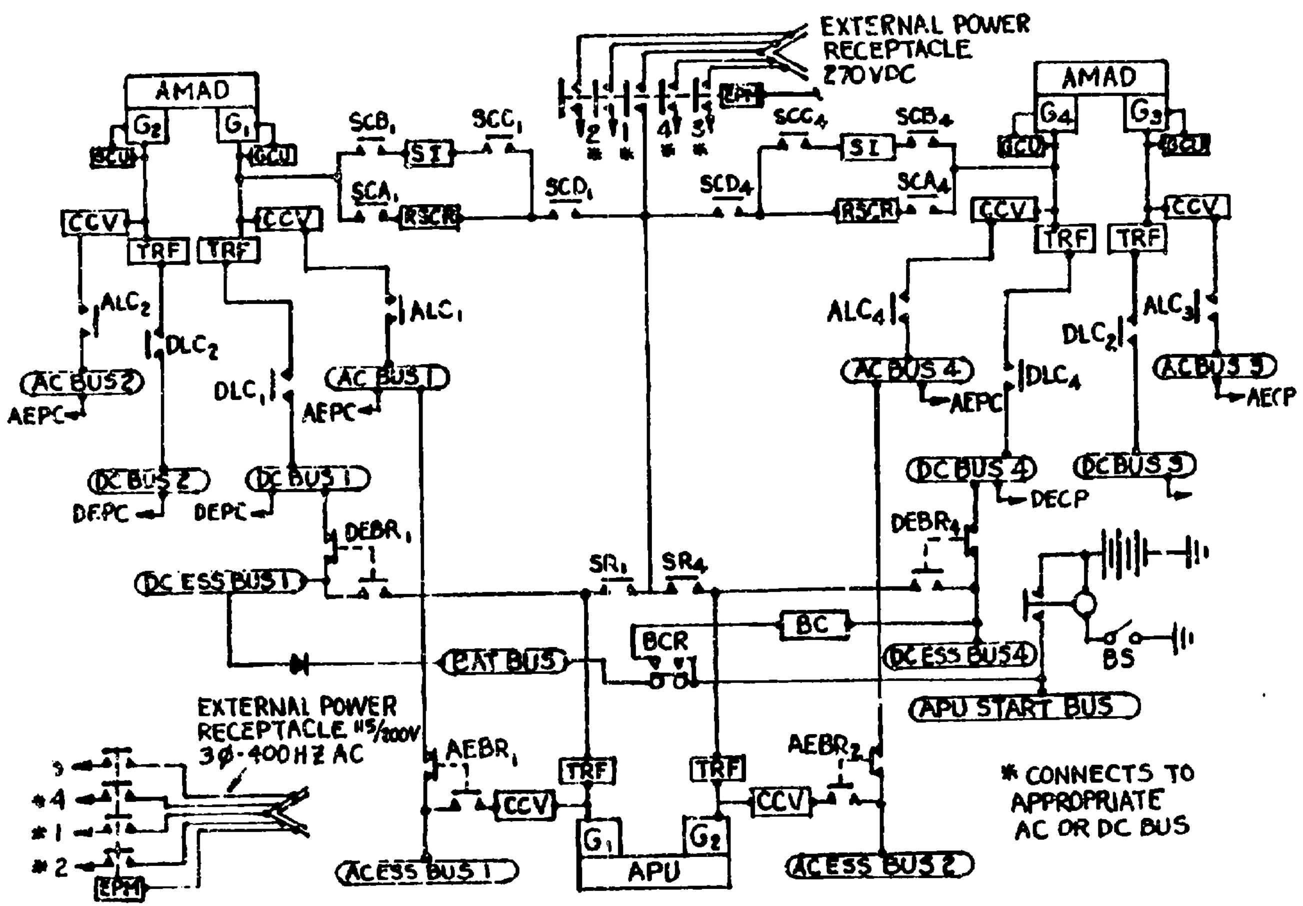


Figure 18. Aircraft I Electrical Power Generation and Distribution System Schematic (Sheet 1 of 2)

GLOSSARY OF ABBREVIATIONS AND ACRONYMS

A

AC - Alternating Current

AEBR - AC Essential Bus Relay

AEPC - AC External Power Contactor

ALC - AC Line Contactor

AMAD - Airframe Mounted Acessory Drive

APU - Auxiliary Power Unit

B

BAT - Battery

BC - Battery Charger

BCR - Battery Charger Relay

BR - Battery Relay

BS - Battery Switch

C

CCV - Cyclo Converter

D

DC - Direct Current

DEBR - DC Essential Bus Relay

DEPC - DC External Power Contactor

DLC - DC Line Contactor

E

EPM - External Power Monitor

ESS. - Essential

EXT. - External

G

G - Generator

GCU - Generator Control Unit

H

HZ - Hertz (cycles per second)

R

RSCR - Reversible Silicon Controlled Rectifier

S

SCA,B,C&D - Starter Contactor A,B,C,D, Etc.

SI - Starter Inverter

SR - Starter Relay

T

TRF - Transformer, Rectifier, Filter

V

VDC - Voltage, DC

Figure 18. Aircraft I Electrical Power Generation and Distribution System Schematic (Sheet 2 of 2)

PTO system power extraction loads could have been balanced using four equal sized generators feeding three systems. However, since the aircraft had two engines this almost inevitably lead to two one generator systems, each generator driven by a different engine, and one two generator system with each of its generators driven by one of the two engines. If it was assumed that all four generators were to be of equal size, for logistics and interchangeability reasons, and were to be of minimum weight, it meant that the two generator system had to have assigned to it twice the continuous load of the other two systems. With the aircraft configuration used for this study, this always led to trouble in the reliability survivability area. In attempting to distribute the loads, to properly load the two generator system, it usually worked out that the engine actuation functions (plug throat, external flaps, etc.) for both engines had to be on the two generator power channel. This meant that a single failure could lead to degrading the whole propulsion system to a marginally fail safe condition. In contrast when each engine was powered by its own system (as is true in the four system approach) only one engine's output was degraded to marginally fail safe after a single failure and it took two failures to achieve the same level of propulsion system degradation. This represented a serious reduction in aircraft reliability and survivability. Attempts to redistribute the loads in other ways, while having each engine's actuation functions powered by separate systems, always seriously impacted the redundancy of power distribution to the flight control system or seriously unbalanced the loads between generators. In either event, a significant negative impact on weight or survibability/reliability always occured when attempting to use a 3 channel power system rather than a four channel approach.

With reference to reason #2 above, it is pointed out elsewhere (see figure 35) that a four signal channel plus model channel fly-by-wire (and light) control was selected for the study aircraft. Using a battery to power the 5th (model) channel, the four power systems approach fitted nicely with the four signal channels and at the same time allowed absolute system separation (power and signal) to be maintained. A three channel power system did not fit so well. At some point a single power system must power two signal channels and thus, a single failure would have eventually lead to the failure of two signal channels.

With reference to reason #3 above, it was found, rather surprisingly, that one of the benefits of a 4 channel dedicated power generation system was that it was significantly smaller and lighter than a conventional 4 channel split-parallel system using bus tie contactors. In the early studies on this program the latter system was thoroughly studied and the results of that study are included in this report for comparison purposes as follows:

> "The electrical power generation and distribution systems (EPGDS), shown schematically in Figure 19 is designed to provide power during conditions of normal and emergency aircraft operation. The aircraft I system, shown in Figure 19 consists of an engine-driven 270 volt DC starter-generating system, and APU-driven backup generator, converted 115/200 volt 400-Hz power, an emergency battery, and provisions for use of an external power source.
>
> The maximum average load demand of Aircraft I, for a 15-minute operating condition, is estimated to be 223 KW. The loads which establish this maximum are those listed in appendix "A" of NA 79-492 (Interim Report Electrical Load Analysis Reference 12) except that items 405 and 406 page A-16 are deleted. The deletion arises from the fact that, unlike the original assumption, the freon compressor is no longer powered electrically but is powered directly by the engine. To supply this mission load the primary electrical system utilizes four 270 volt DC generators, each with a capacity rating of 115 KW. At this rating, the primary electrical power generation system provides a 100 percent

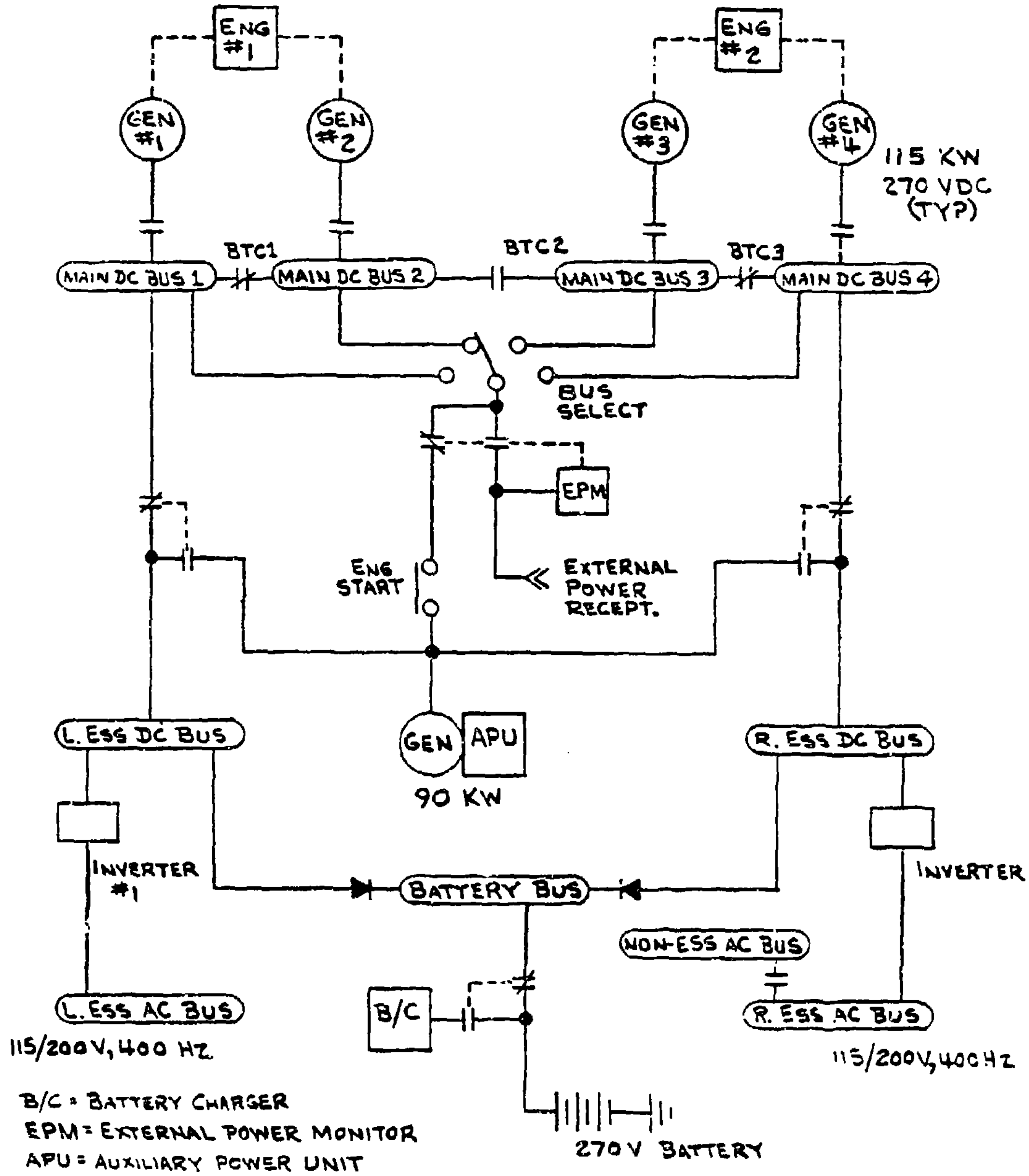


Figure 19. Aircraft I EPGDS Schematic

reserve capacity. The four generators, mounted two-per-engine, supply four main DC buses to support multiredundancy requirements of the flight control system. During normal flight operation bus-tie contactor #2 (BTC 2) is open while BTC 1 and BTC 3 are closed. In this mode the system operates in a split-parallel configuration. Generators #1 and #2 operate in parallel to supply main buses #1 and #2, isolated from main buses #3 and #4, and generators #3 and #4 operate in parallel to supply main buses #3 and #4. In the event of a generator failure, or single engine flameout, BTC 2 will be closed and the remaining generators operated in parallel to ensure an uninterrupted supply of power to the four main DC buses. This arrangement (split/parallel) provides fault isolation by preventing disturbances on one-half of the electrical system from affecting equipment on the other half. In addition to supplying primary electrical power, all 4 generators on aircraft I provide power for starting the main propulsion engines. Aircraft II utilizes a conventioanl pneumatic start system for the engines and the generators supply only primary electrical power.

In configuring the electrical subsystem it is anticipated that 270 HVDC will not always be the most efficient power source. Lighting, instrumentation, avionics, engine controls, and motors (where rapid response is not a critical requirement) are design areas that can fall in this category. A conventional 115/200 volt, 3-phase, 400 Hz electrical conversion system is provided for those subsystems that can conveniently utilize such power. For baseline system sizing considerations all motor loads such as surface control actuators, fuel pumps, ECS fans and pumps, etc., are regarded as powered by 270 HVDC. All housekeeping (non-actuator) loads, except motor loads, are considered to require conventional 400 Hz power. The total 400 Hz load requirement is 44 KW and two 45 KW static inverters provide redundant sources for this amount of power. Of the total capacity available from the 4 generators approximately 110 KW (55 per redundant channel) is allocated for static conversion to a conventional 115/200 volt, 400 Hz power system. This total includes losses in the power conversion devices and provides for a 100-percent reserve capability over estimated load requirements. The remainder of the generating system capacity (175 KW) is reservered for distribution as 270 volt DC power".

From the above discussion of the conventional approach, two critical factors stand out; the size of the generators and the use of 3 bus tie contactors. It will be noted from the above extract that four 115 KW rated generators were going to be used in the conventional system while in the dedicated power channel approach the generator capacity could be reduced to 60/70 KW rated. Since the difference in weight between a 60 and a 115 KW generator of this type was approximately 50 lbs the total weight penalty, associated with the generators, for the conventional system was approximately 200 lbs. It will also be remembered that bus tie contactors were not required in the dedicated power channel approach so an additional weight penalty of 16.5 lb (3 units at 5.5 lb each) would have to be added giving a total weight penalty of 216.5 lb for the use of a conventional system arrangement.

The basic reason that the conventional approach suffered a weight penalty was the fact that, as generators failed, the remaining generators picked up the full system load. After two failures the remaining generators were carrying double their normal continuous load. In contrast the generators in the dedicated approach never carried more than their normal load. In effect there was automatic and weight free load monitoring in the dedicated approach. As generators failed the loads serviced by those generators no longer received power. The inbuilt redundancy of the power using functions, however, insured that the function would continue to operate at its required level of output in the face of the required number of failures.

The conventional system could have reduced its required generator system size, including bus tie contactors, to a weight value approaching that of the dedicated system through the use of a rather elaborate load monitoring system. It was felt, however, that this could only be accomplished at a further reduction in system reliability additive to the relatively poor reliability which already existed in the conventional system due to its propensity for "cascade" type failure. Therefore, consideration of 4 generator/3 power system and/or split bus parallel arrangements were abandoned.

4.1.2.1 Power Generation Definition - Four 60/70 KW generators (two per engine) provided the electrical power required by the study aircraft. As shown in Figure 18, each generator divided its power output to provide both 270 volt DC and 115/200 volt, 400 Hz, 3-phase AC power. The larger portion (approximately 2/3) of each generator system's output was 270 volt DC and was predominantly that power provided for operation of the elctromechanical surface control actuators. To obtain 270 volt DC, part of the wild frequency AC output of the alternator was rectified in the early stages of the VSCF power conditioning.

Conventional 115/200 volt, 400 Hz AC power was provided for avionics and other conventional AC loads such as induction motors (without the use of inverters), lighting, and heating. Although an advanced IDG design offered some advantages over the VSCF system in the areas of size and weight (where all output power was 400 Hz) a VSCF generator was selected because electronic power conversion was more suitable where different types of power output were required and where a majority of the requirement was DC. The generator was a 10-pole, wound rotor unit driven over a speed range of 13750 to 27500 RPM in the power generating mode. Although the load and/or start capabilities differed between generator systems, all four generators (alternator plus cyclo- converter) were designed to be interchangeable for ease of maintainability and inventory preposes. The reason for selection of the generator size and the split between AC and DC loads is discussed at greater length in paragraphs 4.1.4 and 4.1.2.2.

4.1.2.2 270 Volt DC Start Capability - The engine start requirement was 89.5 KW of power applied to the generator output terminals for 35 seconds. To meet this requirement, the generator must be sized as at least a 60 KW unit. At this rating the generator would be capable of providing 90 KW (150-percent) for 2 minutes. The general arrangement of the starting system is shown on Figure 18.

In the starting mode, one synchronous generator on each engine functioned as a brushless DC machine with variable frequency AC power supplied to it. This was made possible throught the use of onboard, dedicated static power inverters ("SI" in Figure 18) to provide programmed voltage and frequency power supply for the starter generators. Each starter-inverter could be powered from an onboard APU or from external power as shown in Figure 18. System operation was analogous to a DC shunt machine supplied by a phase-controlled rectifier. The inverter operated as a phase controlled rectifier to adjust the voltage level and to switch the current among the armature windings. A position sensor on the machine informed the inverter which winding must be supplied with current. The use of the inverters represented a major weight penalty but was more than offset by the elimination of the need for air compressors and air turbine starters such as used on aircraft II (See Figure 45). In addition the need for high power ground air supplies was eliminated since, through the dual functional use of the generator made possible by the inverter, only electrical power was needed for the starting function.

In the start mode, a unique application of the starter-inverter and the 270 VDC rectifier portion of the VSCF power conditioner were combined with a torque converter to crank the engine through an aircraft mounted accessory drive (AMAD) pad.

As shown in Figure 20, upon initiation 270 volt DC power was applied through the 15 KVA inverter to the alternator (as AC power) to bring it up to a programmed synchronous speed of 10,000 RPM. Programmed firing of the inverter transistors was timed, by feeding back rotor speed and position to the inverter, to accelerate the alternator. To protect the system from drawing excessive power from the 270 VDC start bus, the absolute power output from the inverter was sensed and the current limited by adjusting the transistor firing angle. In Figure 18, engine start from the APU was initiated by closing contactors SR1 and SR2. Closure of contactors SCC and SCB routed power through the starter-inverter (SI) to the alternator.

In Figure 20, when the alternator rotor speed (N_2) reached 10,000 RPM the system logic of the controller sent a reversing signal to the SCR's of the 270 VDC full wave bridge rectifier. Concurrently, starter contactors (SC) C and B were opened and starter contactor A was closed. This function transfered the alternator power source from the starter inverter to the reversed SCR's of the 270 volt DC rectifier. The commutation of the main power conditioner SCR's was accomplished by the presence of alternator back EMF. At 10,000 RPM the peak line-to-line voltage of the synchronous motor was momentarily higher than the 270 volt DC bus voltage necessary for SCR commutation. The closure of contactor A also directed the opening of the torque converter fill valve.

At this point, approximately one second transpired since start initiation. The alternator was at full speed, but starter output shaft rotation downstream of the torque converter had not begun. As the torque converter filled, torque from the alternator shaft, controlled to maintain 10,000 RPM, was transferred through the torque converter to the AMAD pad. Only at this time was the full torque requirement of engine starting reflected electrically to the 270 volt DC bus.

Starter output torque followed a curve similar to Figure 21 as the engine accelerated to starter cutout speed. When the AMAD pad speed (N_1) reached starter cutout, the controller logic inhibited the reversing mode signal to the rectifier SCR's and drained the torque converter by removing the fill valve driver signal. As the engine continued its acceleration up to idle speed, and the starter generator tended to slow due to removal of inverter SCR power, the torque flow reversed. Utilizing this torque flow reversal, a set of overrunning clutches was provided which disengaged the torque converter and simultaneously direct coupled the engine to the starter/generator and drove it, as a generator, throughout the engine speed range. The full wave bridge rectifier and filter provided 270 volt DC power for the aircraft primary bus system. As shown in Figure 18, 270 volt DC power and 115/200 volt, 400 Hz AC power were available to the aircraft by closing contactors DLC and ALC.

4.1.2.3 Power Distribution Bus Arrangement - Figure 18 shows the general arrangement of the power distribution system up through the various busses. As previously indicated, the output of the generator was wild frequency AC power. This power was processed in a power conditioner ("PC" in Figure 18) which consisted basically of a voltage regulator and a cyclo converter. The voltage regulator delivered up to 1/3 of the generator's rated capacity, in the form of voltage controlled wild frequency AC power, to the cyclo converter for conversion to constant frequency 400 Hz AC power. The balance of the voltage regulator's output was delivered to a transformer-rectifier-filter unit (TRF" in Figure 18) for conversion to 270 VDC power. The generator control unit ("GCU" in Figure 18) sensed that the generator was out of starting mode, was up to speed and was ready to sustain load. When this occurred it closed the AC and DC contactors.

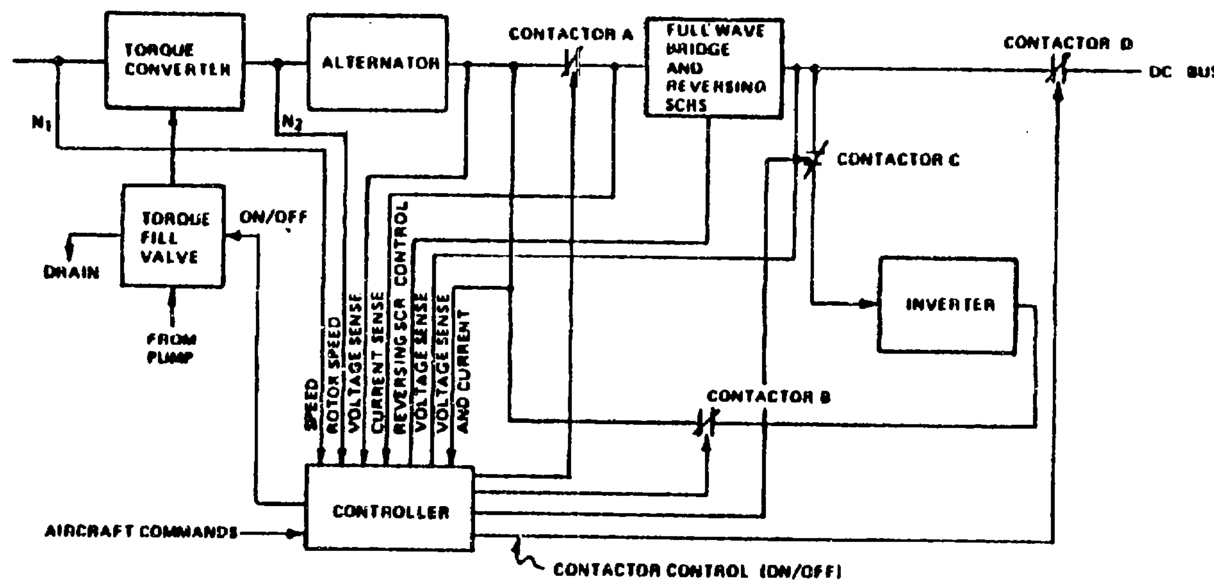


Figure 20. Aircraft I 270 VDC Starting Arrangement.

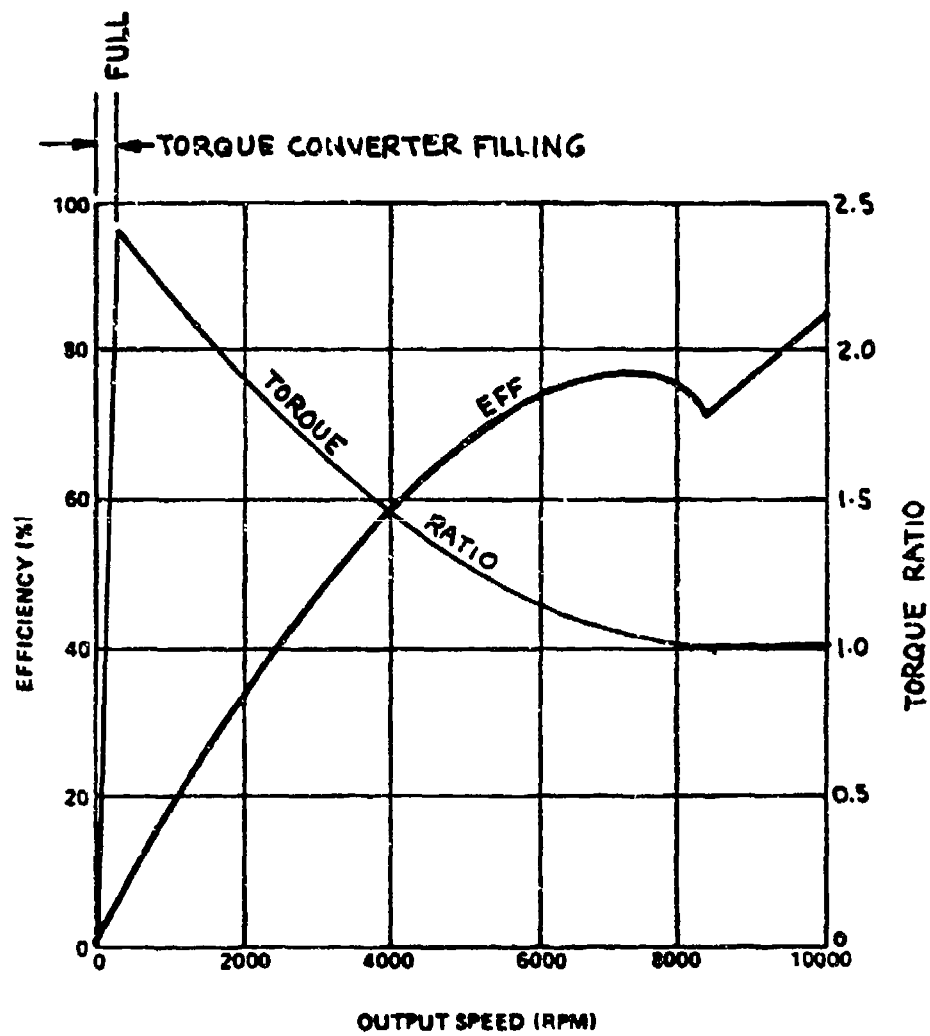


Figure 21. Aircraft I Engine Starting Torque Converter Characteristics

("ALC" and "DLC" for the various systems in Figure 18) and delivered power to all the busses attached to that particular system. From Figure 18 it can be seen that systems No. 2 and No. 3 each powered a primary AC and DC bus while system No. 4 powered an AC and DC essential bus in addition to the primary busses. In common with system No. 4, system No. 1 powered AC and DC primary and essential busses but added to them a battery bus powered through a diode. A large battery was provided. This battery was sufficient to provide a minimum of 4 minutes of power for essential emergency flight control actuation functions as well as for other emergency power requirements (central computer, emergency lighting etc.), occuring at the same time. This battery was charged from system No. 2 via a battery charger and relay (BCR" and "BC" respectively in Figure 18). Normally system No. 1 powered the battery bus and the various emergency functions attached thereto. However, if all systems failed the battery charger relay (BCR) switched to the position shown in Figure 18 and the battery powered the emergency functions. The diode, shown in Figure 18, prevented the battery from delivering power to functions not essential in an emergency such as those attached to the DC essential bus and the primary bus. If power output was re-established, in one or more of the primary or essential busses (say by flying the aircraft down to 20,000 ft and starting the APU) the battery charging relay (BCR) returned to the charging position. The battery also powered the APU start bus. This bus provided power to the APU start motor to bring the APU to self sustaining speed after an engine start had been initiated. The function of the battery switch (BS) and battery relay (BR) was to make it possible to remove all power from all busses while the aircraft was parked and inactive.

The rules governing the assignment of functions to the various busses were as follows:

1. Those functions, and only those function, necessary for recovering from a maneuver and maintaining level or descending flight as well as those functions necessary for towing and parking were assigned to the battery bus.

2. Those functions necessary for a safe return to base and landing were assigned to the various AC or DC essential busses.

3. All other functions were assigned to primary AC or DC busses.

The actual assignment of functions is covered in more detail in paragraph 4.1.4.

As shown in Figure 17 and again in Figure 18 the APU mounted and powered two generators. These were sized based on the starting load requirement discussed in paragraph 4.1.2.2 and were rated at 45 KW each. Since the two generators operated in parallel during engine starts this rating gave a starting system 2 minute rating of 135 KW (45 KW x 2 x 1.5). This would meet the 40 sec start requirement of 90 KW with enough left over to meet essential bus loads of 45 KW. The maximum continuous bus loads which occured on either of the two essential busses during the starting sequence were 34.68 KW (see table 16 sheet 11) and the maximum 5 sec loads were 50.28 KW (see table 16 sheet 12). Based on these figures it was assumed that the maximum 2 minute loads would not exceed 42 KW which was a value within the limits of the 45 KW available.

The APU generators generated and delivered AC and DC power in a manner identical to that already described for the main system generators. Assuming it was designed to start engine No. 1 through generator No. 1 (B1 in Figure 18) the following status would exist initially. In Figure 18 SR_1 and SR_2 would be closed, $AEBR_2$ and $DEBR_2$ would be open on the starter side, AEBR, and DEBR, closed on the starter side and, as already described in paragraph 4.1.2.2, SCB,

and SCC, would be closed while SCD and SCA, would be open. In this way power would be directed to the starter/generator (G1) through the starter inverters (I) to initiate the starting cycle and at the same time power would be directed to AC ESS. bus 1, DC FSS. bus 1 and to the BAT bus. The starting sequence would then proceed as previously described in paragraph 4.1.2.2.

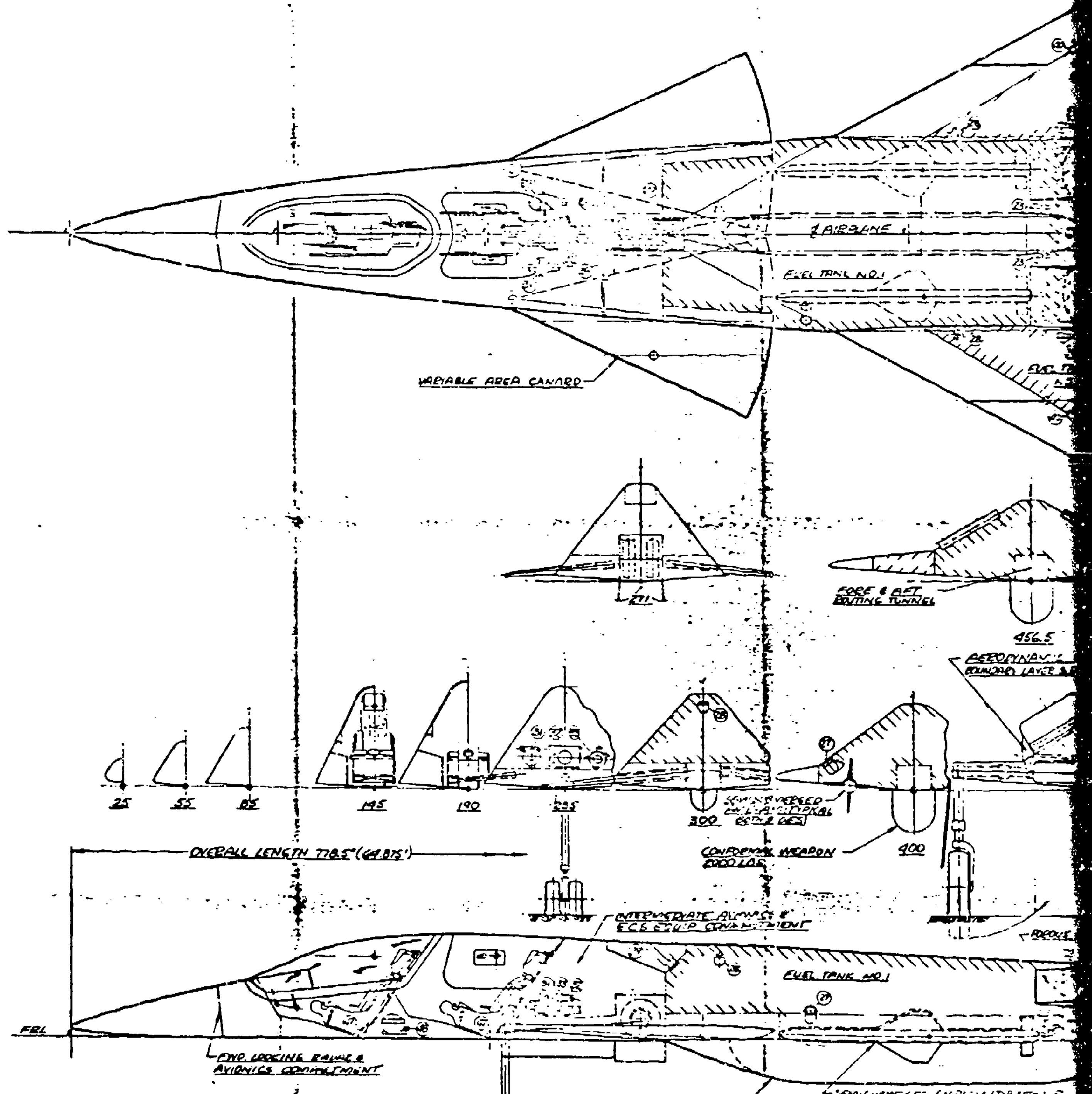
During normal operation the APU would deliver power to the essential busses of system No. 1, or system No. 4 or both dependent upon which system, or systems, had failed. In this instance starter relays SR_1 and SR_2 were open and the APU generators operate independently. The use of two generators in this manner was another attempt to maintain the absolute system separation which was the goal of this system arrangement. While the interlinking involved in the starter system did defeat the absolute purity of system separation sought to a certain extent, it did maintain this separation during all normal and APU powered flight operations. There was also some potential interlinking involved in the battery chargin circuit which could not be avoided. However, even though systems No.1 and No. 2 had some interlinking and thus some possibility of a cascade failure occuring between the two, systems No. 2 and No. 3 maintained their absolute isolation.

4.1.3 Aircraft Inboard Profile - Figure 22 is the inboard profile of the baseline study aircraft. Figure 22 shows the general location of those items which are powered by the electrical system and which are common to study aircraft I and II. Typical of these general locations are the forward, intermediate, and aft avionics compartment and the various fuel pumps. As will be seen elsewhere (paragraph 4.1.1.3.1) it was the general study philosophy that, along with all avionics components, all functions powered by unidirectional motors or nonmodulated bidirectional motors employ conventional 400 Hz AC power. Therefore, these components (avionic "black boxes", fuel pumps, emergency door actuators, radar drives etc.) were considered to be unchanged between aircraft I and aircraft I. and thus did not enter into the trade study except for the impacts their loads had on overall generation system sizing and the constraints their locations placed on wire bundle routing.

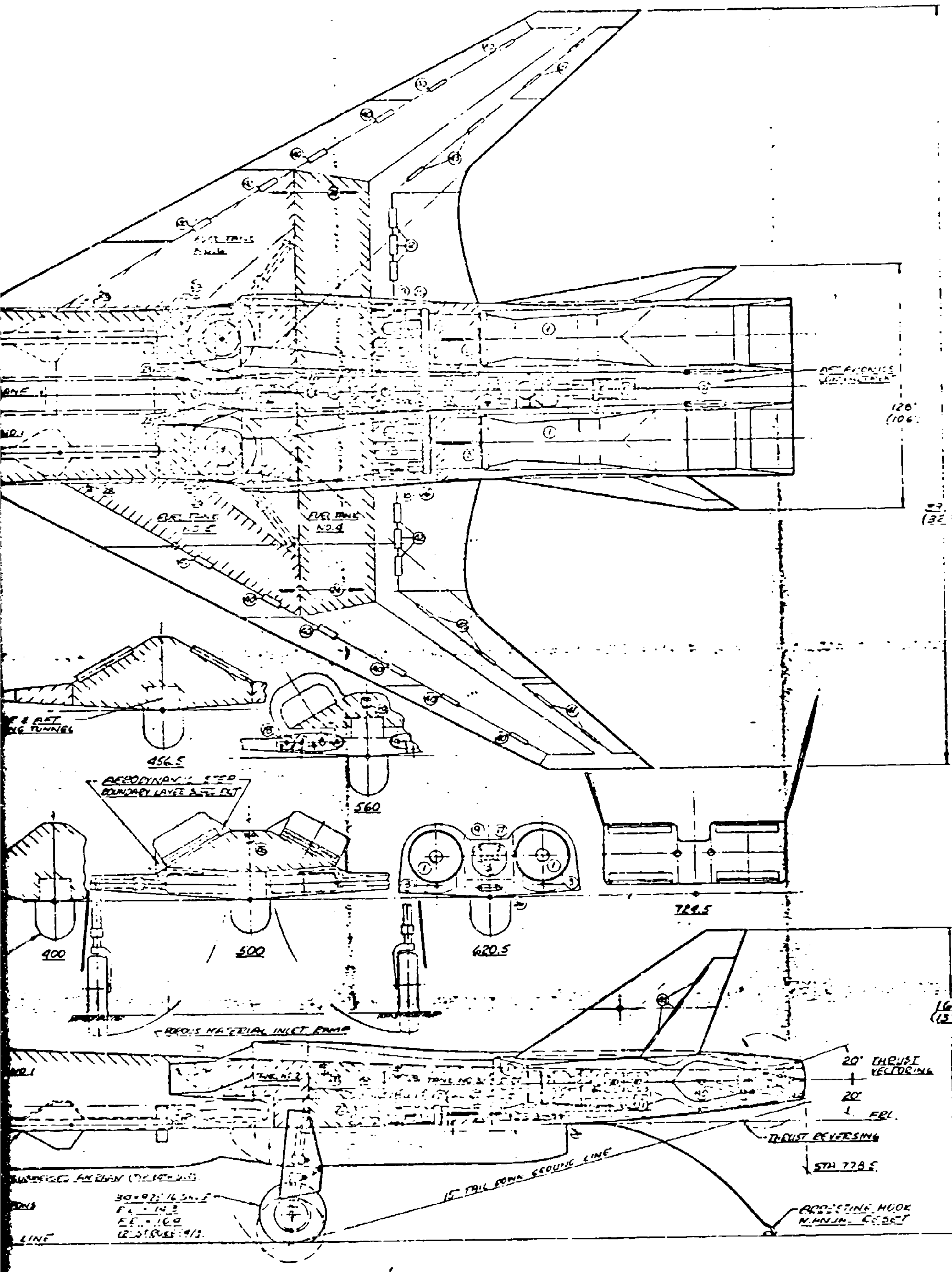
Figure 23 is a specialized version of the inboard profile of figure 22 which provides a schematic of the "all electric" airplane (i.e. aircraft I). As such it defines the relative location of all the major power using equipment items on board the aircraft. Figure 23 also defines the power system or systems servicing each function as well as the general routing of the distribution elements to each function.

Several items stand out in Figure 23. The first is the fact that all four systems run to all parts of the airplane and the second is the rather large volume taken up by inverters, particularly those in the fuel tanks. It had been hoped that only two power supply systems would have to run fore and aft in the fuselage since, with the exception of the rudders, all functions along the fuselage centerline (plug throat, canard, nose gear, etc.) were basically one or two power system units. Had this been possible it would have reduced the wiring system's weight and, to a certain extent, its vulnerability. However, the requirements for load balancing and the need for nose micro-processors (See Figure 37), located in the forward avionics bay, to have at least three independent power sources dictated otherwise. Because there were three large power users in the nose of the aircraft (the gun, the radar, and the defensive subsystems - items 1151, 901, and 1001 repectively on the electrical load analysis) which tended to operate simultaneously during combat, the problem of power balancing these loads, plus the triple power source needs of the micro-processors, dictated that four power supply systems were required in the nose of the aircraft. The same problem was encountered in the aft end of the airplane. Here the high loads were represented by the plug throat and the thrust vector vane. Of the two the thrust vector vane had the highest redundancy requirements (fail operate - fail safe). To meet this requirement, each thrust vector vane needed to have two power supply channels, each of which was preferably powered by its engine. Since there were two engines and two thrust vector vanes, one for each engine, it followed that there was a need for four power supply systems at the aft end of the two engines.

The location of the various inverters required for the various actuation functions can be seen Figure 23. It will be noted that a majority of the inverters, and all of the large liquid cooled types were located in the fuselage. A few of the smaller ambient air cooled units, were located in the wings. The largest inverters were located in the fuselage fuel tanks in the wing carry through area. An end view of these units can be seen in the FS 560 crossection shown in Figure 21. Although the function, size and the weight of the inverters is discussed later in paragraph 4.1.5.1.2, it can be seen that these inverters occupy a considerable volume (41.6 gal) and thus displace a significant weight of fuel (332 lb). Since the total useable fuel capacity of the aircraft was 18,000 lbs this represents a 1.8% fuel capacity reduction and an equivalent range reduction.



VARIABLE AREA CANARD
FUEL TANK NO.1
271
FORE & AFT ROUTING TUNNEL
456.5
25
55
85
145
190
235
300
400
OVERALL LENGTH 778.5" (64.875')
CONFORMAL WEAPON 2000 LBS
INTERMEDIATE AVIONICS & ECS EQUIP COMPARTMENT
FUEL TANK NO 1
FRL
AVIONICS COMPARTMENT
TWO 2000 LB CONFORMAL WEAPONS
WL-60 STATIC GROUND LINE



LIST OF EQ
PROPULSI
1. (2) 199
2. ACCES
3. PTO D
4. AIR TU
5. AUXIL
6. APU
7.
SECOND
8. GENE
9. GENE
10. TRANS
11. INVER
12. VOLTA
13. BATTE
14. POWE
15.
SECOND
16. PRIMA
17. APU
18. PRIMA
19. ENER
20. ACCU
21.
FUEL SY
22. BOOST
23. TRANS
24. EJEC
25. FUEL
26. HEAT
27. GROUN
28. LEVEL
29. INFLI
30.
ENVIRON
31. BOOTS
32. INTER
33. WATE
34. CABIN
35. CABIN
36. PREC
37. PRES
38.
FLIGHT
39. CANA
40. WING
41. AILER
42. ELEV
43. ELEV
44. RUDD
45. ENG
46.
AVIONI
FOR:
INFL
CON
NAV
TARG
DEF
FLIG
FURNI
47. EJE
48. OXY
49. REC

LIST OF EQUIPMENT

PROPULSION SYSTEM

1 (2) 1995 ADVANCED VARIABLE CYCLE ENGINE
2 ACCESSORY DRIVE GEARBOX
3 PTO DRIVE SHAFT
4 AIR TURBINE STARTER
5 AUXILIARY POWER UNIT
6 APU LOX TANK
7

SECONDARY POWER SYSTEM (ELECTRICAL)

8 GENERATOR 50 KVA 115/200 VAC, 150F (2 REQD)
9 GENERATOR 7 KVA
10 TRANSFORMER RECTIFIER UNIT 200 AMP (2 REQD)
11 INVERTER 1500 VA 1φ
12 VOLTAGE REGULATOR (APU SYS.)
13 BATTERIES 40 AMP HR NI-CAD (2 REQD.)
14 POWER CONTACTOR (2 REQD.)
15

SECONDARY POWER SYSTEM (HYDRAULIC 8000 PSI)

16 PRIMARY PUMPS 28 GPM (4 REQD.)
17 APU PUMP 28 GPM
18 PRIMARY RESERVOIR
19 EMERGENCY RESERVOIR
20 ACCUMULATOR
21

FUEL SYSTEM

22 BOOSTER PUMPS (2 REQD.)
23 TRANSFER PUMPS (10 REQD.)
24 EJECTOR PUMPS (4 REQD.)
25 FUEL/OIL HEAT EXCHANGER (GEN & ADG OIL)
26 HEAT EXCHANGER PUMP
27 GROUND REFUEL RECEPTACLE
28 LEVEL CONTROL VALVES (6 REQD.)
29 INFLIGHT REFUEL RECEPTACLE
30

ENVIRONMENTAL CONTROL SYSTEM

31 BOOTSTRAP TURBO COMPRESSOR
32 INTERCOOLER HEAT EXCHANGER
33 WATER SEPARATOR
34 CABIN SAFETY VALVE
35 CABIN PRESSURE REGULATOR
36 PRECOOLER & EJECTOR
37 PRESSURE REGULATORS (ENG. MTD) NOT SHOWN
38

FLIGHT CONTROL SYSTEM

39 CANARD ACTUATORS, TANDEM UNBALANCED, 2 REQ'D.
40 WING L.E. DEVICE, MECH. HINGE ROTARY ACTUATORS 12 REQ'D
41 AILERONS, MECHANICAL " " " 4 REQ'D.
42 ELEVONS INBOARD MECH " " " 6 REQ'D.
43 ELEVONS OUTBOARD " " " " 4 REQ'D.
44 RUDDER/SPEED BRAKE, MECH. HINGE ROTARY ACT. 6 REQ'D.
45 ENGINE THRUST VECTOR ACTUATORS, TANDEM UNBALANCED, 2 REQ'D.
46

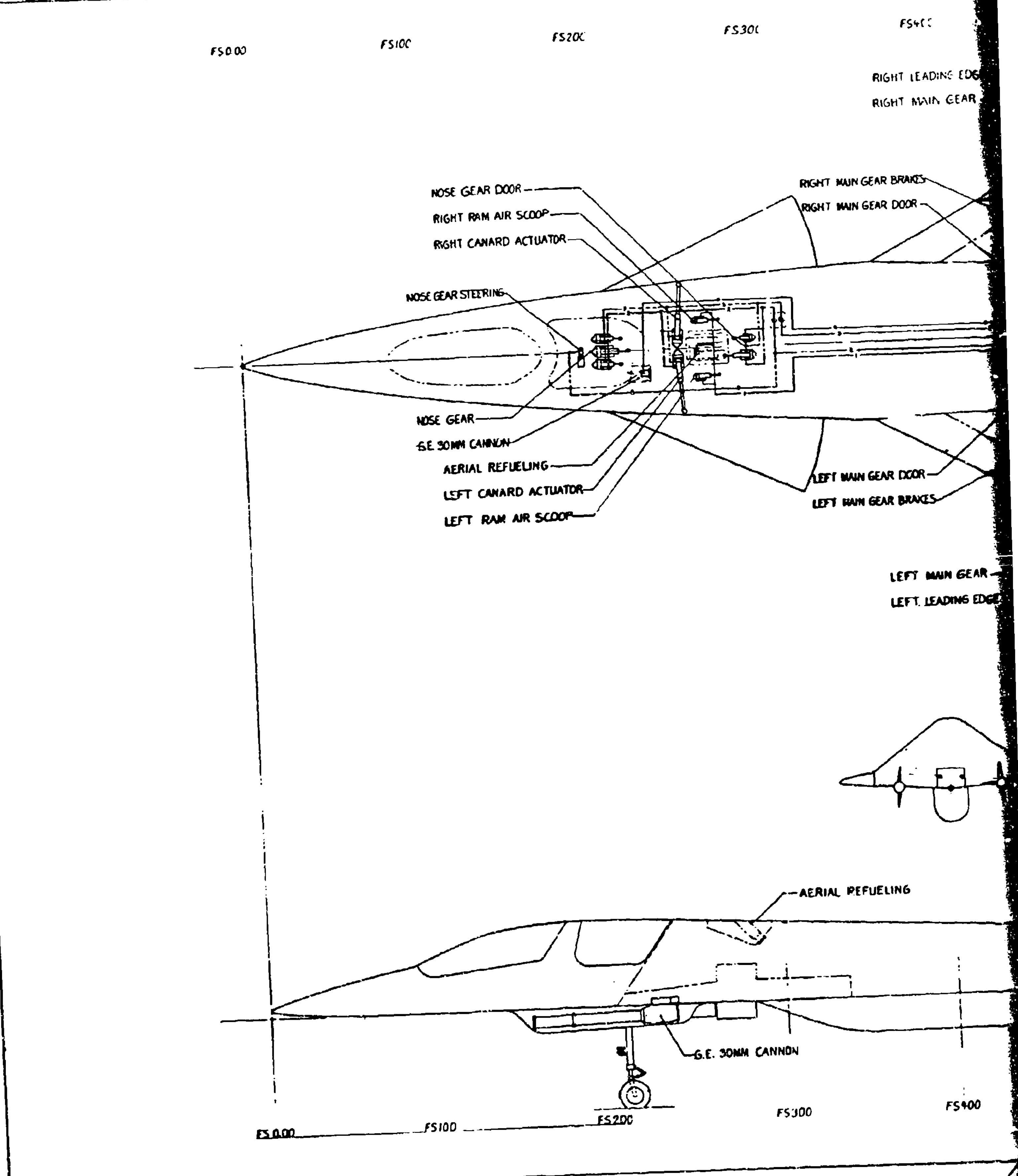
AVIONICS SYSTEM 922 LBS; 50 FT³ SPACE PROVISIONS FOR:

INFORMATION MANAGEMENT SYS.
COMMUNICATIONS SYS
NAVIGATION SYS
TARGET ACQUISITION SYS.
DEFENSIVE SYS.
FLIGHT CONTROL SYS.

FURNISHINGS

47 EJECTION SEATS, STENCEL S-III S 2 REQ'D.
48 OXYGEN SYSTEM - 5 LITER LOX
49 RELIEF PROVISIONS - HORN & BOTTLE (NOT SHOWN)

Figure 22. ATS Inboard Profile



FS0.00
FS100
FS200
FS300
RIGHT LEADING EDGE
RIGHT MAIN GEAR
NOSE GEAR DOOR
RIGHT RAM AIR SCOOP
RIGHT CANARD ACTUATOR
NOSE GEAR STEERING
RIGHT MAIN GEAR BRAKES
RIGHT MAIN GEAR DOOR
NOSE GEAR
G.E. 30MM CANNON
AERIAL REFUELING
LEFT CANARD ACTUATOR
LEFT RAM AIR SCOOP
LEFT MAIN GEAR DOOR
LEFT MAIN GEAR BRAKES
LEFT MAIN GEAR
LEFT LEADING EDGE
AERIAL REFUELING
G.E. 30MM CANNON
FS0.00
FS100
FS200
FS300
FS400

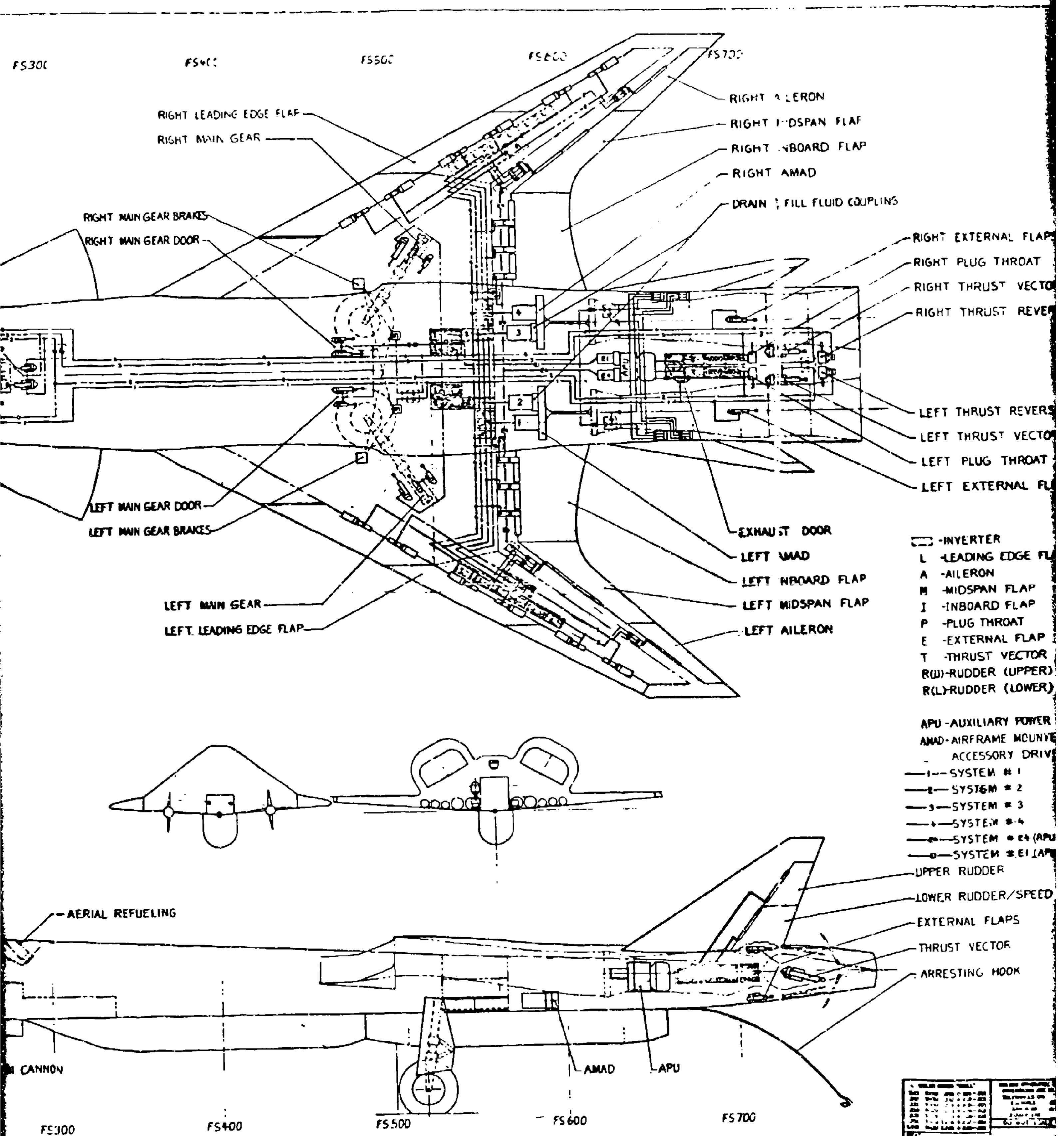


Figure 23. Electric Actuatio

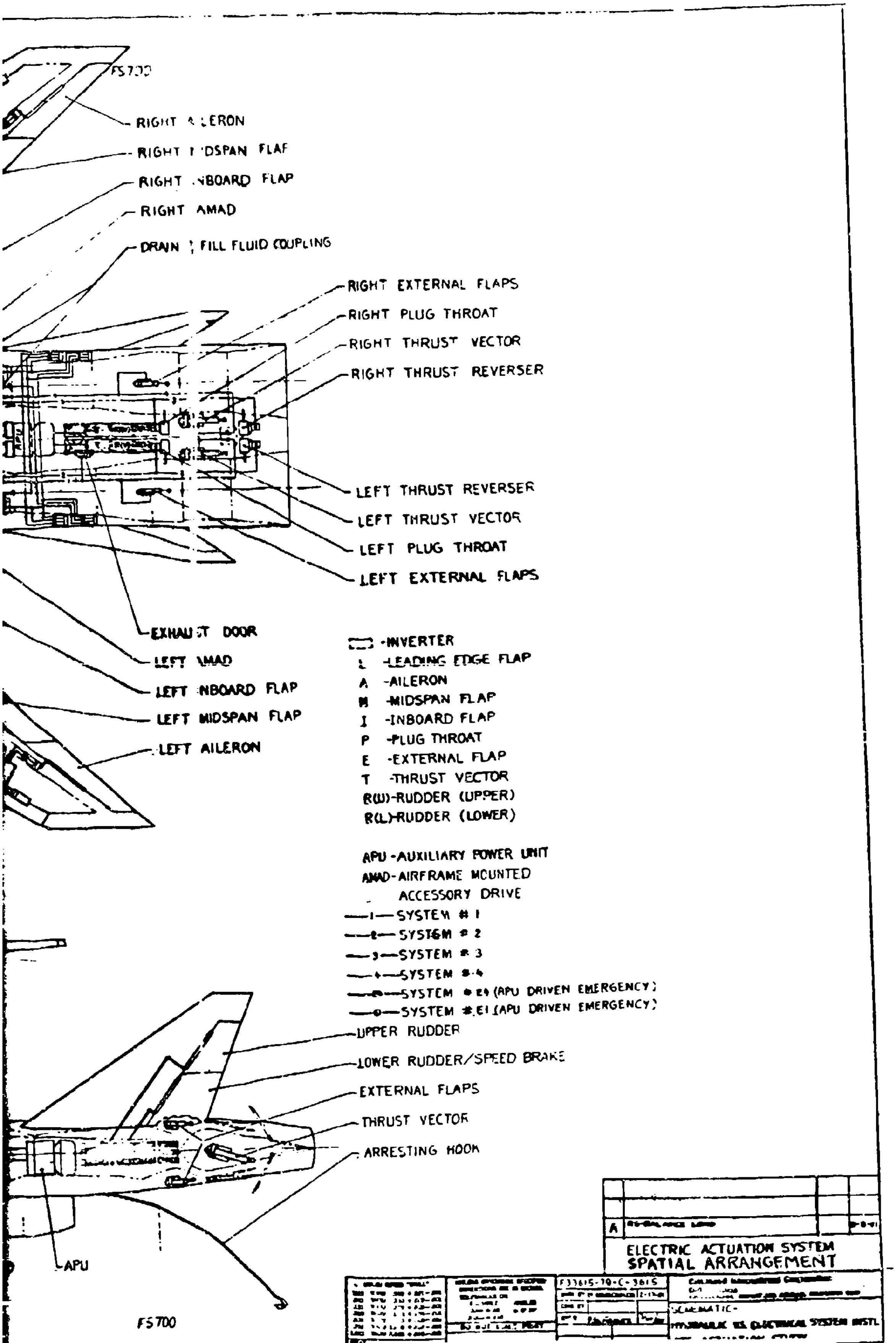


Figure 23. Electric Actuation System Spatial Arrangement.

4.1.4 Electrical System Load Analysis - An electrical load analysis was included as a part of the first interim technical report (Reference 8). Although this load analysis was roughly representative of the loads seen by the electrical system, several changes in system philosophy and refinements in system definition occured after the load analysis was originally issued. These changes were of sufficient magnitude, and had a sufficient impact on the loads reflected back to the generators, that it was felt, that not only was a revision to the load analysis needed, but a new approach to the presentation of the loads was required. The more important of these changes are listed as follows:

1. Revised flight control actuation requirements as discussed in paragraph 2.3.1.5.

2. Revised utility actuator requirements as discussed in paragraph 2.3.1.

3. The deletion of the ground cooling fuel heat sink door as discussed in paragraph 2.3.1.1.

4. The deletion of the canopy as an actuation function per paragragh 2.3.1.2.

5. The revision in approach to utility engine actuation functions as outlined in paragraph 2.3.1.3.

6. The deletion of major portions of the environmental control system (ECS) as a trade study item (See paragraph 2.3.1.4).

7. The adoption of the four independent channel approach to electrical power distribution disscussed in paragraph 4.1.2.

The last item (Item 7) was the major determinant of the need for a new method of presenting electrical loads for the purposes of this study. The computerized load analysis system used at Rockwell was eminently satisfactory for presenting and summing the loads on a conventional electrical system using bus tie contactors (i.e. where any pair of generators can eventually see the loads normally carried by four generators). However, the load analysis technique, as constitued at the time of this report, did not gracefully handle the identification and apportioning of various loads among the various busses, the effects of load monitoring, nor the impact of four dedicated systems (as opposed to a bus tie contactor system). Any one or all of these items could have been handled by the computerized load analysis system through extensive revision, however, it was not felt that the required expenditure of time nor the increased complexity of the final readout justified the effort required for this program. Therefore, a revised approach using manual tabulation was devised. The results are shown in Tables 15, 16 (12 pages) and 17. Table 15 is an

TABLE 15. SUPPLEMENTAL ELECTRICAL LOAD ANALYSIS ITEM NUMBER BREAKDOWN

1111	INBOARD TRAILING EDGE FLAP
1112	MIDSPAN TRAILING EDGE FLAP
1113	OUTBOARD TRAILING EDGE FLAP (AILERON)
1114	UPPER RUDDER
1115	LOWER RUDDER
1116	LEADING EDGE FLAP
1117	CANARD
1118	THRUST VECTOR VANE
1121	EXTERNAL FLAP
1122	PLUG THROAT
1123	THRUST REVERSER
1131	NOSE GEAR
1132	NOSE GEAR DOORS
1133	MAIN GEAR
1134	MAIN GEAR DOORS
1135	NOSE GEAR STEERING
1136	MAIN GEAR BRAKES
1141	RAM AIR SCOOP (RIGHT)
1142	RAM AIR SCOOP (LEFT)
1143	EXHAUST DOOR-EQUIPMENT BAY
1144	GROUND COOLING FUEL HEAT SINK
1151	30 MM GUN DRIVE
1161	REFUEL RECEPTACLE
1162	APU START MOTOR

TABLE 16. COMBAT - 5 SEC LOADS 4 CHANNELS OPERATIVE (SHEET 1 OF 12)

ITEM NO	115/200 V 400 HZ AC BUSSES 1	2	3	4	E 1	E 4	270 VDC BUSSES 1	2	3	4	E 1	E 4	BAT
101	✓0.216												
102				✓0.216									
103	✓0.054												
104													
105	✓0.243												
106				✓0.243									
107	✓0.297												
108				✓0.297									
109	✓0.243												
110				✓0.243									
111					✓								
201	✓0.837												
202	0.270			0.270	✓	✓							
203	0.810												✓
204				✓0.810									
302	0.054			0.054	✓	✓							
303	0.027			0.027	✓	✓							
304	0.054			0.054	✓	✓							
305	0.027			0.027	✓	✓							
306	0.027			0.027	✓	✓							
314	0.027				✓								
315				0.027		✓							
316	0.027				✓								
317				0.027		✓							
318				0.027		✓							
401	✓3.220			✓3.220									
403	✓0.870			✓0.870									
407	✓0.054			✓0.054									
501	✓	✓											
502	✓	✓											
503	✓	✓											
504	✓	✓											
505	✓	✓											
506	✓	✓											
FG TOTAL	7.357	0	0	6.547	—	—	0	-	0	0	—	—	—

COMBAT - 5 SEC LOADS
4 CHANNELS OPERATIVE

TABLE 16. COMBAT - 5 SEC LOADS 4 CHANNELS OPERATIVE (SHEET 2 OF 12)

ITEM	115/200 V 400 HZ AC BUSSES						270 VDC BUSSES						
NO	1	2	3	4	E 1	E 4	1	2	3	4	E 1	E 4	BAT
509	✓												
510		✓											
511	✓												
512		✓											
513	✓												
514		✓											
515	✓												
516	✓0.135												
517		✓0.108											
518													✓
601	5.913				✓								
602				5.913		✓							
603	1.863				✓								
604				1.863		✓							
605	0.999				✓								
606				0.999		✓							
607	0.999				✓								
608				0.999		✓							
609					✓								
610						✓							
701							0.486			0.243		✓	✓
702	✓0.837												
703							1.296			0.649		✓	✓
801							0.270			0.135		✓	✓
802							0.108			0.054		✓	✓
803							0.065			0.063		✓	✓
804							0.040			[illegible]		✓	✓
901			✓12.744										
902			✓1.755										
903			✓0.162										
1001		✓19.791											
1002			✓1.701										
PG. TOTAL	10.746	19.899	16.362	9.774	—	—	2.269	0	0	[illegible]	—	—	—

COMBAT - 5 SEC LOADS
4 CHANNELS OPERATIVE

ITEM NO	115/200 V 400 HZ AC BUSSES 1	2	3	4	E1	E4	270 VDC BUSSES 1	2	3	4	E1	E4	BAT
1111							17.722	8.861	8.861	17.722			✓
1112							3.544	1.772	1.772	3.544			✓
1113							0.708	0.708	0.708	0.708	✓	✓	
1114							0.912	0.456	0.456	0.912			✓
1115							0.866	0.433	0.433	0.866			✓
1116						*	3.315	3.315	3.315	3.315	✓	✓	
1117								0.283	0.283				
1118							1.200			1.200			
1121							13.767			13.767			
1122								42.918	42.918				
1123								28.613	28.613				
1131		✓	✓										
1132		✓	✓										
1133			✓										
1134			✓										
1135								✓					
1136												✓	✓
1141													✓
1142													✓
1143													✓
1144	✓												
1151										* 43.86			
1161	✓												
1162													✓
TOTAL THIS PAGE							42.034	87.359	87.359	[illegible]			
* CORRECTED TOTAL THIS PAGE							28.023	58.239	58.239	57.267			
ALL PAGE TOTAL	18.103	19.899	16.362	16.321			30.291	58.239	58.239	53.455			

COMBAT - 5 SEC LOADS
4 CHANNELS OPERATIVE

*ACTUAL ACTUATION POWER CORRECTED TO 2/3 OF SUMMED TOTAL BASED ON THE ASSUMPTION THAT ALL ACTUATION FUNCTIONS WILL NOT PEAK SIMULTANEOUSLY

TABLE 16. COMBAT - 5 SEC EMERGENCY 2 CHANNELS OPERATIVE (SHEET 4 OF 12)

ITEM NO	115/200 V 400 HZ AC BUSSES 1	2	3	4	E1	E4	270 VDC BUSSES 1	2	3	4	E1	E4	BAT
1111							31.837	✓ 39.000 ▲	✓	✓			✓
1112							6.380	✓ 2.658	✓	✓			✓
1113							1.417	✓ 1.417	✓		✓	✓	
1114							1.366	✓ 0.683	✓	✓			
1115							1.300	✓ 0.650	✓	✓			✓
1116						*	6.630	✓ 5.630	✓ *		✓	✓	
1117								✓ 0.566	✓				
1118							✓ 2.400			✓			
1121							✓ 13.767			✓			
1122								✓ 42.916	✓				
1123								✓ 23.613	✓				
1131		✓	✓										
1132		✓	✓										
1133			✓										
1134			✓										
1135								✓					
1136												✓	✓
1141													✓
1142													✓
1143													✓
1144	✓												
1151										✓			
1161	✓												
1162													✓
TOTAL THIS PAGE							55.097	123.135					
* CORRECTED TOTAL THIS PAGE							43.39?	95.000 ← $\frac{2}{3}(123.135-39)+39$					
ALL PAGE TOTAL	18.103	19.999					45.64?	95.000					

COMBAT - 5 SEC. EMERGENCY

2 CHANNELS OPERATIVE

▲ ASSUMES SURFACE POWERED BY SINGLE ACTUATOR APPROACHES STALL FOR 5 SECONDS

* ACTUAL ACTUATION POWER CORRECTED TO 2/3 OF SUMMED TOTAL BASED ON THE ASSUMPTION THAT ALL ACTUATION FUNCTIONS WILL NOT PEAK SIMULTANEOUSLY

TABLE 16. COMBAT - CONTINUOUS 4 CHANNELS OPERATIVE (SHEET 5 OF 12)

ITEM NO	115/200V 400HZ AC BUSSES 1	2	3	4	E1	E4	270VDC BUSSES 1	2	3	4	E1	E4	BAT
101	✓0.216												
102				✓0.216									
103	✓0.054												
104				✓0.054									
105	✓0.243												
106				✓0.243									
107	✓0.297												
108				✓0.297									
109	✓0.243												
110				✓0.243									
111					✓								
201	✓0.837												
202	0.270			0.270	✓	✓							
203	0.810												✓
204				✓0.810									
302	0.054			0.054	✓	✓							
303	0.027			0.027	✓	✓							
304	0.054			0.054	✓	✓							
305	0.027			0.027	✓	✓							
306	0.027			0.027	✓	✓							
314	0.027				✓								
315				0.027		✓							
316	0.027				✓								
317				0.027		✓							
318				0.027		✓							
401	✓2.310			✓2.310									
403	✓0.520			✓0.520									
407	✓0.054			✓0.054									
501	✓	✓											
502	✓	✓											
503	✓	✓											
504	✓	✓											
505	✓	✓											
506	✓	✓											
PG TOTAL	6.097	0	0	5.197									

COMBAT-CONTINUOUS
4 CHANNELS OPERATIVE

TABLE 16. COMBAT - CONTINUOUS 4 CHANNELS OPERATIVE (SHEET 6 OF 12)

ITEM	115/200 V 400HZ AC BUSSES						270 VDC BUSSES						
NO	1	2	3	4	E 1	E 4	1	2	3	4	E 1	E 4	BAT
509	✓												
510		✓											
511	✓												
512		✓											
513	✓												
514		✓											
515	✓												
516	✓ 0.135												
517		✓ 0.108											
518													✓
601	2.133				✓								
602				2.133		✓							
603	0.675				✓								
604				0.675		✓							
605	0.297				✓								
606				0.297		✓							
607	0.297				✓								
608				0.297		✓							
609					✓								
610						✓							
701							0.486			0.243		✓	✓
702	✓ 0.837												
703							1.296			0.648		✓	✓
801							0.270			0.135		✓	✓
802							0.108			0.054		✓	✓
803							0.068			0.068		✓	✓
804							0.040			0.040		✓	✓
901			✓ 12.744										
902			✓ 1.755										
303			✓ 0.162										
1001		✓ 19.791											
1002			✓ 1.701										
PG. TOTAL	4.336	19.899	16.362	3.424	—	—	2.268	0	0	1.185	—	—	—

COMBAT - CONTINUOUS
4 CHANNELS OPERATIVE

TABLE 16. COMBAT - 5 CONTINUOUS 4 CHANNELS OPERATIVE (SHEET 7 OF 12)

ITEM	115/200V 400HZ AC BUSSES						270 VDC BUSSES						
NO	1	2	3	4	E1	E4	1	2	3	4	E1	E4	BAT
1111							8.861	4.431	4.431	8.861			✓
1112							1.772	0.886	0.886	1.772			✓
1113							0.354	0.354	0.354	0.354	✓	✓	
1114							0.456	0.228	0.228	0.456			✓
1115							0.433	0.217	0.217	0.433			✓
1116							1.823	1.823	1.823	1.823	✓	✓	
1117								0.142	0.142				
1118							0.600			0.600			
1121							6.884			6.884			
1122								21.459	21.459				
1123													
1131		✓	✓										
1132		✓	✓										
1133			✓										
1134			✓										
1135								✓					
1136												✓	✓
1141													✓
1142													✓
1143													✓
1144	✓												
1151										✓			
1161	✓												
1162		□ APU	START	BUS									✓□
TOTAL THIS PAGE							21.183	29.540	29.540	21.183			
CORRECTED TOTAL THIS PAGE													
ALL PAGE TOTAL	10.493	19.899	16.362	8.621			23.415	29.540	29.540	22.371			

COMBAT- CONTINUOUS
4 CHANNELS OPERATIVE

TABLE 16. COMBAT - CONTINUOUS EMERGENCY 2 CHANNELS OPERATIVE (SHEET 8 OF 12)

ITEM	115/200V 400HZ AC BUSSES						270 VDC BUSSES						
NO	1	2	3	4	E 1	E 4	1	2	3	4	E 1	E 4	BAT
1111							✓15.919	✓6.615	✓	✓			✓
1112							3.190	✓1.329	✓	✓			✓
1113							0.709	✓0.709	✓		✓	✓	
1114							0.683	✓0.342	✓	✓			✓
1115							0.650	✓0.325	✓	✓			✓
1116							3.315	✓3.315	✓		✓	✓	
1117								✓0.283	✓				
1118							✓1.200			✓			
1121							✓6.884			✓			
1122								✓21.549	✓				
1123								✓	✓				
1131		✓	✓				COMBAT - CONTINUOUS EMERG.						
1132		✓	✓				2 CHANNELS OPERATIVE						
1133			✓										
1134			✓										
1135								✓					
1136												✓	✓
1141													✓
1142													✓
1143													✓
1144	✓												
1151										✓			
1161	✓												
1162		□ APU START BUS											✓□
TOTAL THIS PAGE							32.550	34.377					
CORRECTED TOTAL THIS PAGE													
ALL PAGE TOTAL	19.103	19.959					32.550	34.377					

TABLE 16. COMBAT - CONTINUOUS EMERGENCY APU CHANNELS 1 AND 4 OPERATIVE
(SHEET 9 OF 12)

ITEM NO	115/200 V 400 HZ AC BUSSES 1	2	3	4	E1	E4	270 VDC BUSSES 1	2	3	4	E1	E4	BAT
1111							13.212			13.212			
1112							2.659			2.659			
1113							.709			.709			
1114							.684			.684			
1115							.650			.650			
1116							3.315			3.315			
1117													
1118													
1121													
1122							COMBAT - CONTINUOUS EMERGENCY APU CHANNELS 1 AND 4 OPERATIVE						
1123													
1131													
1132													
1133													
1134													
1135													
1136													
1141													
1142													
1143													
1144													
1151													
1161													
1162													
TOTAL THIS PAGE							21.229			21.229			
CORRECTED TOTAL THIS PAGE							21.229			21.229			
							+2.268			+1.195			
ALL PAGE TOTAL	4.725			4.752			23.497			22.417			
				SYSTEM TOTAL			28.222			27.165			

TABLE 16. COMBAT - CONTINUOUS EMERGENCY APU CHANNELS 1 AND 4 OPERATIVE
(SHEET 10 OF 12)

ITEM NO	115/200 V 400HZ AC BUSSES						270 VDC BUSSES						
	1	2	3	4	E 1	E 4	1	2	3	4	E 1	E 4	BAT
1111							26.583			26.583			
1112							5.317			5.317			
1113							1.417			1.417			
1114							1.367			1.367			
1115							1.300			1.300			
1116							6.630			6.630			
1117													
1118													
1121													
1122							COMBAT - CONTINUOUS EMERGENCY APU CHANNELS 1 AND 4 OPERATIVE						
1123													
1131													
1132													
1133													
1134													
1135													
1136													
1141													
1142													
1143													
1144													
1151													
1161													
1162													
TOTAL THIS PAGE							40.614			40.614			
CORRECTED TOTAL THIS PAGE							28.409			28.409			
ALL PAGE TOTAL	11.097		11.124				30.677			24.597			
				SYSTEM TOTAL			41.774			40.721			

TABLE 16. COMBAT - CONTINUOUS EMERGENCY APU CHANNEL 1 OPERATIVE
(SHEET 11 OF 12)

ITEM NO	115/200 V 400 HZ AC BUSSES 1	2	3	4	E 1	E 4	270 VDC BUSSES 1	2	3	4	E 1	E 4	BAT
1111							18.608						
1112							3.722						
1113							.709						
1114							.684						
1115							.650						
1116							3.915						
1117													
1118													
1121													
1122							COMBAT - CONTINUOUS EMERGENCY APU CHANNEL 1 OPERATIVE						
1123													
1131													
1132													
1133													
1134													
1135													
1136													
1141													
1142													
1143													
1144													
1151													
1161													
1162													
TOTAL THIS PAGE							27.688						
CORRECTED TOTAL THIS PAGE							+2.268 29.954						
ALL PAGE TOTAL	4.725					SYSTEM TOTAL	+4.725 34.681						

TABLE 16. COMBAT - CONTINUOUS EMERGENCY APU CHANNEL 1 OPERATIVE (SHEET 12 OF 12)

ITEM NO	115/200 V 400 HZ AC BUSSES 1	2	3	4	E 1	E 4	270 VDC BUSSES 1	2	3	4	E 1	E 4	BAT
1111							37.216						
1112							7.444						
1113							1.417						
1114							1.367						
1115							1.300						
1116							6.630						
1117													
1118													
1121													
1122													
1123													
1131													
1132													
1133													
1134													
1135													
1136													
1141													
1142													
1143													
1144													
1151													
1161													
1162													
TOTAL THIS PAGE							55.374						
CORRECTED TOTAL THIS PAGE							36.916 +2.268						
ALL PAGE TOTAL	11.097						39.184						
				SYSTEM TOTAL			50.281						

COMBAT - CONTINUOUS EMERGENCY APU CHANNEL 1 OPERATIVE

extension, and further breakdown, of the item number breakdown used in the original load analysis (reference 8, Appendix A pages A-11, A-16, A-21, A-26, and A-31). Basically Table 15 is a functional breakdown of item 1100 on page A-31 of Appendix A. Table 15 assigns an item number to each function in the eleven hundred series instead of breaking down the actuation functions by mission segments as was done on page A-31. This brought the eleven hundred series breakdown in line with that used for the rest of the original electrical load analysis.

Table 16 itemizes the loads occuring during combat for the 5 second and continuous load condition. The combat mission segment was selected for detailed breakdown because it represented the highest loads imposed on the generator and on the AMAD and engine PTO system. The first column on Table 16 lists the functional item numbers as taken from Appendix A plus the revised breakdown of the eleven hundred series.

The next thirteen columns in Table 16 list the various busses used in the 400 Hz AC and 270 VDC systems of aircraft I. They match the busses shown in the aircraft I power generation and distribution system schematic of Figure 16. The bus (or busses) to which the functional load, represented by an item number, was attached was indicated by a check mark. The magnitude of the load was entered as a numerical value in the column representing the AC or DC system (No 1, 2, 3 or 4) in which the load ultimately appeared. The location of the check mark and the load numerical value did not necessarily coincide. This occured only when the load was attached to the system's primary bus. Where no numerical load value was entered for a particular item number in the table, it indicated that no power was provided to that function during the mission segment under consideration (combat) or that the time under load was so short that it appeared only under 5 sec. loads but not under continuous loads (item 1151 - 30 MM gun drive for example). Sheets 1, 2 and 3 of Table 16 tabulate the loads appearing on the AC and DC busses of power systems 1, 2, 3, and 4 under the "combat 5 sec. load" condition with 4 channels (systems) operative. Combat 5 sec. loads were a basis for determining the overload requirements of the generator system. Sheet 4 of Table 16 shows the "combat 5 sec. emergency loads" with only 2 systems operative. Sheets 5, 6 and 7 show the "combat-continuous load" condition with 4 channels (systems) operative. Combat-continuous loads were the basis for determining the basic rating requirements of the generator system. Sheet 8 of Table 16 shows the "combat-continuous emergency loads" with only 2 systems operative. In both the case of sheet 4 and sheet 8 "emergency" loads, the loads for item numbers less than item 1111 were not included because the loads for these lesser item numbers were identical to the values already listed on sheets 1 and 2 (vis à vis page 4) and sheets 5 and 6 (vis à vis sheet 8). The item loads for each bus were totaled sheet by sheet for each condition (5 second or continuous) and a grand total was accumulated for each bus on the third sheet. It will be noted, on sheet 3 and 4 on Table 16, that only 2/3 of the actuation loads were used for determining the total load for the 5 second condition. This was in consonance with the ground rule established in paragraph 2.3.4.

In addition to load determinations, the level of power source redundancy could be approximated from Table 16. The number of columns in which load entries (or check marks) appear for a given item number indicate the level of power source redundancy for that function. As an example of the extremes of power source redundancy which were incorporated in the actuation functions of aircraft I, consider the inboard flaps (item 1111) and the main landing gear (item 1133). The inboard flaps had access to 5 power sources (4 generators plus a battery) while the main gear had access to only one. Actually, when considering primary power sources, the disparity between the two was not as great as it would at first appear. The mid span flaps had access to 3 primary power sources (2 engines and a battery) while the main gear had access to 2 (an engine and free fall). In both instances the APU was not considered a primary power source because it could only be started below 20,000 feet.

It will be noted in Table 16 that the loads on an individual operating system increase as other systems become inoperative. As an example, consider DC busses No. 1 and No. 2 on sheets 3 and 4 respectively of Table 16. Sheet 3 sums the DC loads for all four busses at the bottom of the sheet (all page total) for the condition where all 4 channels (systems) are fully operative. Sheet 4 sums the loads in a similar manner for systems No. 1 and No. 2 with the assumption that systems No. 3 and No. 4 have failed. This set of circumstances could occur, as an example, if high altitude battle damage had been experienced in which system No. 3 and No. 4 had been wiped out, and in which violent evasive maneuvers were in progress, and the APU could not be started because of the high altitude. Table 17 provides a summation of the AC, DC, and total loads accumulated on sheets 1 through 8 of Table 16. It can be seen from Table 17 that the generator size was determined from the loads on power system (channel) No. 2 under the headings "combat continuous emergency loads" and "combat 5 sec. load emergency" with 2 channels operative". From the summation it can be seen that the selected generator ratings were satisfactory. While the continuous rating of 70 KW selected for the generator left an apparent 29% margin for growth, the 120 KW 5 sec rating had a much smaller 4.4% margin but was still satisfactory. (See paragraph 4.1.6.)

The loads used in the preparation of Table 16 were taken from two sources. The first source was the electrical load analysis, reference 8, Appendix A pages 1 through 46. This was used for determining all loads associated with functions through item No. 1002. The second source was the utility and flight control function loads included as Figure 7 and Figure 8 in this report. These tables give output loads at the surface or function being powered. In order to convert these output loads to loads at the generator terminals, it was necessary to provide system loss data. This loss data for flight control functions is provided by Figure 24. The figure shows that the losses were broken down into three major categories; (1) power hinge losses, (2) motor losses, and (3) distribution losses. As will be seen later (paragraph 4.1.5) these losses, in terms of percent rated loads, were nearly identical for all

TABLE 17. AIRCRAFT I - ELECTRICAL SYSTEM LOAD SUMMATION

POWER DESCRIPTION	POWER SYSTEM NUMBER			
	1	2	3	4
COMBAT CONTINUOUS LOADS - 4 CHANNELS OPERATIVE				
AC POWER (KW)	10.493	19.899	16.362	8.621
DC POWER (KW)	23.415	29.540	29.540	22.371
TOTAL SYSTEM POWER (KW)	33.908	49.439	45.902	30.992
COMBAT CONT. EMERGENCY LOADS - 2 CHANNELS OPERATIVE				
AC POWER (KW)	18.103	19.899		
DC POWER (KW)	32.550	34.377		
TOTAL SYSTEM POWER (KW)	50.653	54.276		
COMBAT 5 SEC. LOADS - 4 CHANNELS OPERATIVE				
AC POWER (KW)	18.103	19.899	16.362	16.362
DC POWER (KW)	30.219	58.239	58.239	58.455
TOTAL SYSTEM POWER (KW)	48.394	78.138	74.601	74.266
COMBAT 5 SEC. LOAD EMERGENCY - 2 CHANNELS OPERATIVE				
AC POWER (KW)	18.103	19.899		
DC POWER (KW)	45.666	95.090		
TOTAL SYSTEM POWER (KW)	69.769	114.989		

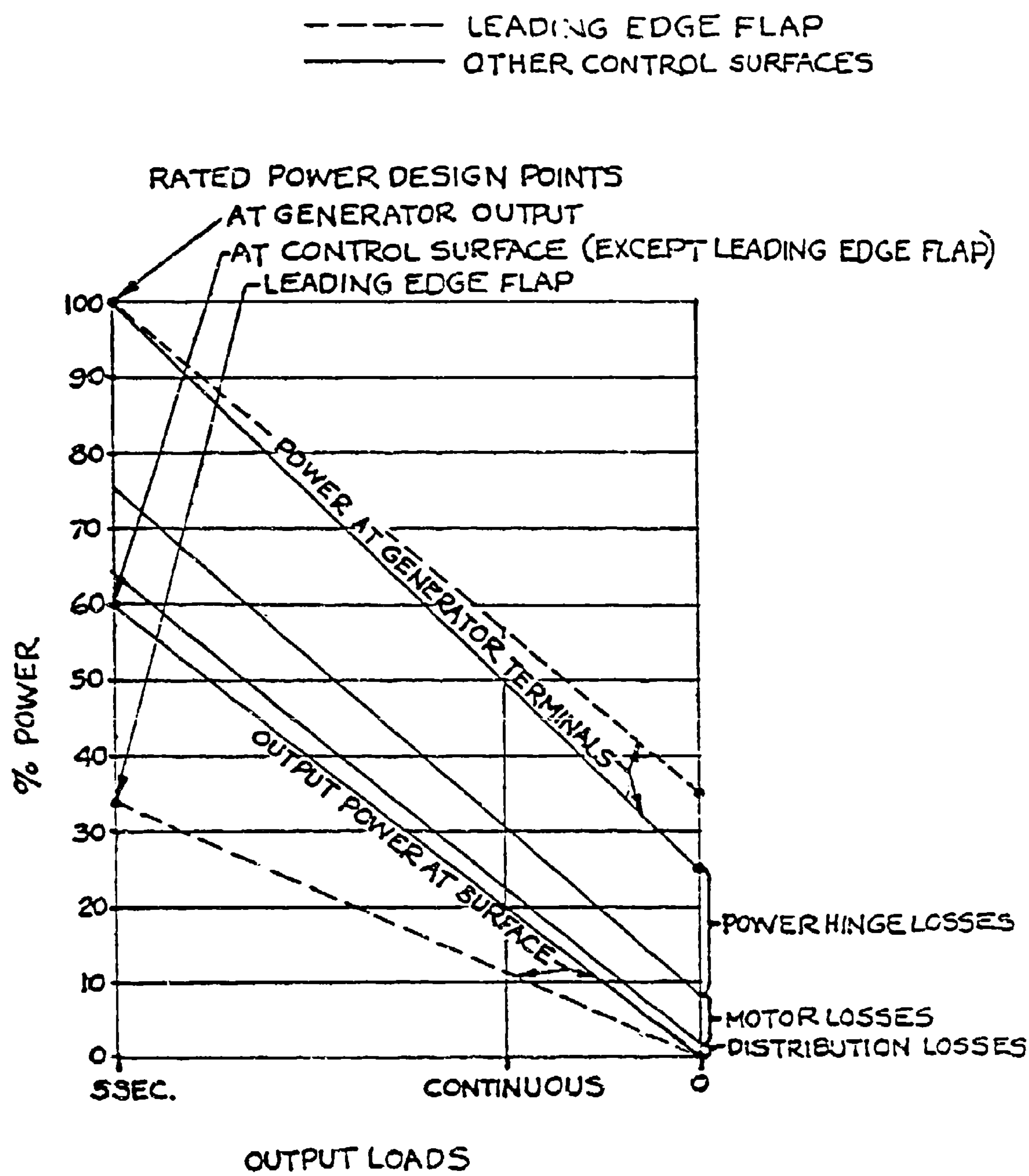


Figure 24. Flight Control Actuator Load Characteristics During Combat.

flight control actuation functions except the leading edge flap. The losses in the leading edge flap system were purposely made high to give it "no back" characteristics in the face of a power system failure. Because the losses in most of the flight control system functions were nearly identical (and for this study were treated as being identical) they were plotted by category as the solid lines in Figure 24. Only the limits for the leading edge flap function, i.e. the output power at the surface and the generator terminal power, were plotted in Figure 24 to avoid complicating the figure.

The power hinge losses, as used in Figure 24, consisted of the losses in the power hinge itself (or, in the case of the canard function, in the ball screw) and the losses in the gearhead. The motor losses consisted of the losses in the motor itself plus the losses in the inverter. The distribution losses consisted of all losses in the distribution system between the generator terminals and the inverter terminals. It can be seen in Figure 24 that the typical flight control actuation function was 60% power efficient at rated conditions and that the leading edge flap system was only 34% efficient. The individual efficiencies which provided the basis for these values are discussed later in more detail (paragraph 4.1.5). It can also be seen in Figure 24 that the power required at the generator terminals for "continuous" operation was 50% (57% for the leading edge flaps) of that required at rated power design conditions (5 sec loads). It is also interesting to note that, even with no load on the output, 25% of rated power is required (36% for the leading edge flaps) at the generator terminals to achieve rated rates of motion. The data from Figure 24 was used to assist in determining the load entries for item numbers 1111 through 1123 in Table 16.

Another factor which had to be considered in generating the flight control function entries in Table 16 was the fact that, when more than one actuator was powering a surface or control function, the actuators shared load and the individual actuator loads were reduced. In contrast, when various systems were rendered inoperative to the point that only one actuator powered a particular surface, or control function, that actuator attempted to carry full load but could only do so up to the limits of its design rated load limit capability. Calculations were made based on design rated (5 sec load) conditions to take this into account. These calculations are as follows:

5 SECOND LOAD ACTUATOR POWER COMPUTATIONS

P_{DS} = Design power per surface, at design load and rate, required at surface

P_{DA} = Design power per actuator, at design load and rate, required at surface

P_{GS} = Power required at generator terminals per surface

P_{GA} = Power required at generator terminals per actuator

R_A = Percent of design power P_{DS} or P_{GS} required per actuator x 0.01

η_H = Efficiency - motor mounting interface to surface being actuated

η_M = Efficiency - motor at design power

η_D = Efficiency - distribution system at design power

$$P_{GS} = \frac{P_{DS}}{\eta_H \times \eta_M \times \eta_D} \qquad P_{GA} = P_{GS} \times R_A$$

P_{GAP} = Power required per actuator at generator terminals when operated in parallel with other actuators on the same surface.

N_{AS} = Number of actuators operating per surface

$$P_{GAP} = \frac{P_{GS}}{N_{AS}}$$

DESIGN POINT POWER INPUT (5 SEC) LOADS

INBOARD FLAP (ITEM NO. 1111)

P_{DS} = 15.95 KW (1) $\qquad \eta_M$ = 0.850

R_A = 0.70 (2) $\qquad \eta_D$ = 0.939

η_H = 0.752 (3)

$$P_{GS} = \frac{15.950}{0.752 \times 0.850 \times 0.939} = 26.583 \text{ KW} \qquad P_{GA} = 26.546 \times 0.70 = 18.608 \text{ KW}$$

$$P_{GAP} = \frac{26.583}{3} = 8.861 \text{ KW} \quad (N_{AS} = 3)$$

$$P_{GAP} = \frac{26.583}{2} = 13.229 \text{ KW} \quad (N_{AS} = 2)$$

$$P_{GAP} = P_{GA} = 18.608 \text{ KW} \quad (N_{AS} = 1)$$

MIDSPAN FLAP (ITEM NO. 1112)

P_{DS} = 3.19 KW (1) $\qquad \eta_M$ = 0.580

R_A = 0.70 (2) $\qquad \eta_D$ = 0.940

η_H = .752 (3)

$$P_{GS} = \frac{3.190}{0.600} = 5.317 \text{ KW} \qquad P_{GA} = 5.317 \times 0.70 = 3.722 \text{ KW}$$

$$P_{GAP} = \frac{5.317}{3} = 1.772 \text{ KW} \quad (N_{AS} = 3)$$

$$P_{GAP} = \frac{5.317}{2} = 2.658 \text{ KW} \quad (N_{AS} = 2)$$

$$P_{GAP} = P_{GA} = 3.722 \text{ KW} \quad (N_{AS} = 1)$$

OUTBOARD TRAILING EDGE (ITEM NO. 1113)

$P_{DS} = 0.85$ KW (1) $\quad \eta_M = 0.850$

$R_A = 1.00$ (2) $\quad \eta_D = 0.939$

$\eta_H = .752$ (3)

$$P_{GS} = \frac{0.850}{0.600} = 1.417 \text{ KW} \qquad P_{GA} = 1.417 \times 1.00 = 1.417 \text{ KW}$$

$$P_{GAP} = \frac{1.417}{2} = .708 \text{ KW} \quad (N_{AS} = 2)$$

$$P_{GAP} = P_{GA} = 1.417 \text{ KW} \quad (N_{AS} = 1)$$

UPPER RUDDER (ITEM NO. 1114)

$P_{DS} = 0.82$ KW (1) $\quad \eta_M = 0.850$

$R_A = 0.50$ (2) $\quad \eta_D = 0.939$

$\eta_H = 0.752$ (3)

$$P_{GS} = \frac{0.820}{0.600} = 1.367 \text{ KW} \qquad P_{GA} = 1.367 \times 0.500 = 0.683 \text{ KW}$$

$$P_{GAP} = \frac{1.367}{3} = 0.456 \text{ KW} \quad (N_{AF} = 3)$$

$$P_{GAP} = \frac{1.367}{2} = 0.683 \text{ KW} \quad (N_{AF} = 2)$$

$$P_{GAP} = P_{GA} = 0.683 \quad (N_{AF} = 1)$$

LOWER RUDDER (ITEM NO. 1115)

P_{DS} = 0.780 KW (1) $\quad \eta_M$ = 0.850

R_A = 0.500 (2) $\quad \eta_D$ = 0.939

η_H = 0.752 (3)

$$P_{GS} = \frac{0.780}{.600} = 1.300 \text{ KW} \qquad P_{GA} = 1.300 \times 0.500 = .650 \text{ KW}$$

$$P_{GAP} = \frac{1.300}{3} = 0.433 \text{ KW} \quad (N_{AF} = 3)$$

$$P_{GAP} = \frac{1.300}{2} = 0.650 \text{ KW} \quad (N_{AF} = 2)$$

$$P_{GAP} = P_{GA} = 0.650 \text{ KW} \quad (N_{AF} = 1)$$

LEADING EDGE FLAP (ITEM NO. 1116)

P_{DS} = 2.24/3 = 0.747 KW (1) $\quad \eta_M$ = 0.850

R_A = 1.000 (2) $\quad \eta_D$ = 0.939

η_H = 0.424 (3)

$$P_{GS} = \frac{0.747}{0.424 \times 0.850 \times 0.939} = 2.210 \text{ KW} \qquad P_{GA} = 2.210 \times 1.000 = 2.210 \text{ KW}$$

$$P_{GAP} = \frac{2.210}{2} = 1.105 \text{ KW} \quad (N_{AF} = 2)$$

$$P_{GAP} = P_{GA} = 2.210 \text{ KW} \quad (N_{AF} = 1)$$

CANARD (ITEM NO. 1117)

$$P_{DS} = 0.170 \text{ KW } (1) \qquad \eta_M = 0.850$$

$$R_A = 1.000 \ (2) \qquad \eta_D = 0.940$$

$$\eta_H = 0.752 \ (3)$$

$$P_{GS} = \frac{0.170}{0.600} = 0.283 \text{ KW} \qquad P_{GA} = 0.283 \times 1.000 = 0.283 \text{ KW}$$

$$P_{GAP} = \frac{0.283}{2} = 0.142 \text{ KW} \qquad (N_{AF} = 2)$$

$$P_{GAP} = P_{GA} = 0.283 \text{ KW} \qquad (N_{AF} = 1)$$

THRUST VECTOR VANE (ITEM NO. 1118)

$$P_{DS} = 0.72 \text{ KW } (1) \qquad \eta_M = 0.850$$

$$R_A = 1.000 \ (2) \qquad \eta_D = 0.939$$

$$\eta_H = 0.752 \ (3)$$

$$P_{GS} = \frac{0.720}{0.600} = 1.200 \text{ KW} \qquad P_{GA} = 1.200 \times 1.000 = 1.200 \text{ KW}$$

$$P_{GAP} = \frac{1.200}{2} = 0.600 \text{ KW} \qquad (N_{AF} = 2)$$

$$P_{GAP} = P_{GA} = 1.200 \text{ KW} \qquad (N_{AF} = 1)$$

EXTERNAL FLAP (ITEM NO. 1121)

P_{DS} = 4.130 KW (4)

R_A = 1.000

η_H = 0.752 (3)

η_M = 0.850

η_D = 0.939

$$P_{GS} = \frac{4.130}{0.600} = 6.883 \text{ KW}$$

$$P_{GA} = 6.883 \times 1.000 = 6.883 \text{ KW}$$

P_{GAP} = P_{GA} = 6.883 KW (N_{AF} = 1)

PLUG THROAT (ITEM NO. 1122)

P_{DS} = 25.520 KW (4)

R_A = 1.000

η_H = 0.745 (5)

η_M = 0.850

η_D = 0.939

$$P_{GS} = \frac{25.520}{0.745 \times 0.850 \times 0.939} = 42.918 \text{ KW}$$

$$P_{GA} = 42.918 \times 1.000 = 42.918 \text{ KW}$$

P_{GAP} = P_{GA} = 42.918 KW

THRUST REVERSER (ITEM NO. 1123)

P_{DS} = 19.640 KW (4)

R_A = 1.000

η_H = 0.860 (5)

η_M = 0.850

η_D = 0.939

$$P_{GS} = \frac{19.640}{0.860 \times 0.850 \times 0.939} = 28.613 \text{ KW}$$

$$P_{GA} = 28.613 \times 1.000 = 28.613 \text{ KW}$$

$$P_{GAP} = P_{GA} = 28.613 \text{ KW}$$

(1) See Table 8

(2) See Table 11

(3) See Tables 18 and 19

(4) See Table 9

(5) See Table 9

Another factor considered in making the load entries in Table 16 was the fact that the flight control functions (item 1111 through 1118) and the engine (flight control type) functions (items 1121 through 1123) were motor loads. This meant that, when approaching an output stall condition, these functions could increase the apparent load at the generator terminals by nearly a factor of 2.5. This phenomena is discussed in more detail in paragraph 4.1.5.1.1. To account for this, the load analysis of Table 16 assumed that, during an emergency (5 sec loads), at least one actuator approached stall for a short period of time in recovering from the maneuver which the aircraft found itself in at the time of the emergency. This is illustrated in Table 16 page 4.

Four additional pages (sheets 9 through 12) were added to the Table 16 load analysis. These were added to cover emergency loads experienced during APU operation and were used to help size the APU and the two APU generators. The sizing of the APU generators has already been discussed in paragraph 4.1.2.3 and, as pointed out, the sizing used the data from Table 16 sheets 11 and 12. Sheets 9 and 10 of Table 16 were included to provide the data for sizing the APU itself. The maximum load on the APU was that resulting from operating two systems simultaneously or the sum of the two generator loads. From sheet 9 it can be seen that the maximum continuous APU load requirement was 28.222 KW + 27.169 KW = 55.391 KW (74.25H.P.) and from sheet 10 it can be seen that the maximum design load (5 sec) was 41.774 KW + 40.721 KW = 82.495 KW (110.58 H.P.).

4.1.5 Power Utilization - For the purpose of this portion of the actuation trade study the utilization functions on the aircraft were divided into three general categories as follows:

1. Flight control actuation

2. Utility actuation

3. Other power consuming systems

The first two categories had a major impact on all aspects of the trade study while the impact of the third category was largely confined to it's affects on generator sizing.

4.1.5.1 Flight Control Actuation - The design and definition of the major flight control actuators was subcontracted to Airesearch because of their extensive experience in the development of electro mechanical actuation systems (Reference 17 and 20). Using the flight control actuation requirements listed in Table 8 Airesearch submitted a comprehensive set of preliminary design parameters. These are shown as Tables 18, 19, and 20. Airesearch also submitted envelope and weight data which are shown in Figures 25 through 31. It can be seen that both the inboard flap actuation system (Figure 25) and the airleron actuation system (Figure 27) met the envelope requirements established in Figures 14 and 15 respectively even though the fit in both instances was very tight.

TABLE 18. APPROXIMATE DRIVE SIZING

Actuator	Stall Load	Ratio*	η_{GR}*	Drives per Surface	Type** Diameter x Length Weight
Inboard Flap	151,000 in-lb	1320	0.80	1-3	Epicyclic 3.5 x 9.4 in 11.1 lb
Midspan Flap	30,000 in-lb	1320	0.80	1-3	Epicyclic 1.6 x 9.1 in 2.9 lb
Aileron	26,000 in-lb	2640	0.80	1-2	Epicyclic 1.4 x 11.8 in 3.2 lb
Upper Rudder	9,000 in-lb	3300	0.80	1-3	Epicyclic 1.4 x 4.1 in 1.1 lb
Lower Rudder	13,000 in-lb	1650	0.80	1-3	Epicyclic 1.6 x 4.0 in 1.3 lb
L.E. Flap	175,000 in-lb	13,200	0.45	1-2	Epicyclic 4.0 x 7.2 in 12.8 lb
Canard	2,400 lb	1.25×10^{-2} in-rev^{-1}	0.80	1-2	Ball Screw*** 1.0(2.0) x 19.9 in 11.6 lb

* Overall, using motor data from calc. No. 1-002. See attached procedure

** Selected to meet envelope & performance *** ID (OD) x L

TABLE 19. APPROXIMATE GEARHEAD SIZING

Actuator	Input Stall Torque	Ratio*	η_{GH}*	Qty per Surface	Diameter x Length Weight **
Inboard Flap	141.8 in-lb	~90	0.94	3	~5.5 x 5.0 in ~15.0 lb
Midspan Flap	28.4 in-lb	~90	0.94	3	3.0 x 3.5 in 6.0 lb
Aileron	12.6 in-lb	~180	0.94	2	2.5 x 3.0 in 2.5 lb
Upper Rudder	3.2 in-lb	~220	0.94	3	1.5 x 1.8 in 1.0 lb
Lower Rudder	6.3 in-lb	~110	0.94	3	2.0 x 2.0 in 2.0 lb
L.E. Flap	15.8 in-lb	~90	0.94	2	2.5 x 3.0 in 3.0 lb
Canard	3.2 in-lb	~20	0.94	2	1.5 x 1.8 in 1.0 lb

* For gearhead only. Sized by use of data from Air Report No. 80-17284

** Selected to satisfy input torque requirements

TABLE 20. APPROXIMATE MOTOR SIZING*

Actuator	Calc'd Power	Motor Power	η_{GR}	Motors per Surface	Speed** Diameter x Length Weight
Inboard Flap	44.93 hp 33.52 kw	45.0 hp 33.6 kw	0.80	3	20,000 rpm 3.5 X 10.6 in 12 lb
Midspan Flap	8.93 hp 6.66 kw	9.0 hp 6.7 kw	0.80	3	20,000 rpm 2.3 x 7.7 in 2.3 lb
Aileron	3.87 hp 2.89 kw	4.0 hp 3.0 kw	0.80	2	20,000 rpm 1.9 x 6.7 in 1.0 lb
Upper Rudder	1.07 hp 0.80 kw	1.0 hp 0.75 kw	0.80	3	20,000 rpm 1.9 x 5.1 in 0.3 lb
Lower Rudder	1.55 hp 1.15 kw	2.0 hp 1.5 kw	0.80	3	10,000 rpm 2.3 x 6.0 in 2.4 lb
L.E. Flap	5.15 hp 3.84 kw	5.0 hp 3.7 kw	0.45	2	10,000 rpm 2.3 x 8.1 in 2.6 lb
Canard	0.94 hp 0.70 kw	1.0 hp 0.75 kw	0.80	2	10,000 rpm 2.3 x 5.3 in 0.5 lb

* Assumes drive efficiency of η_{gr}. Sized by attached procedure

** Selected to meet envelope & performance

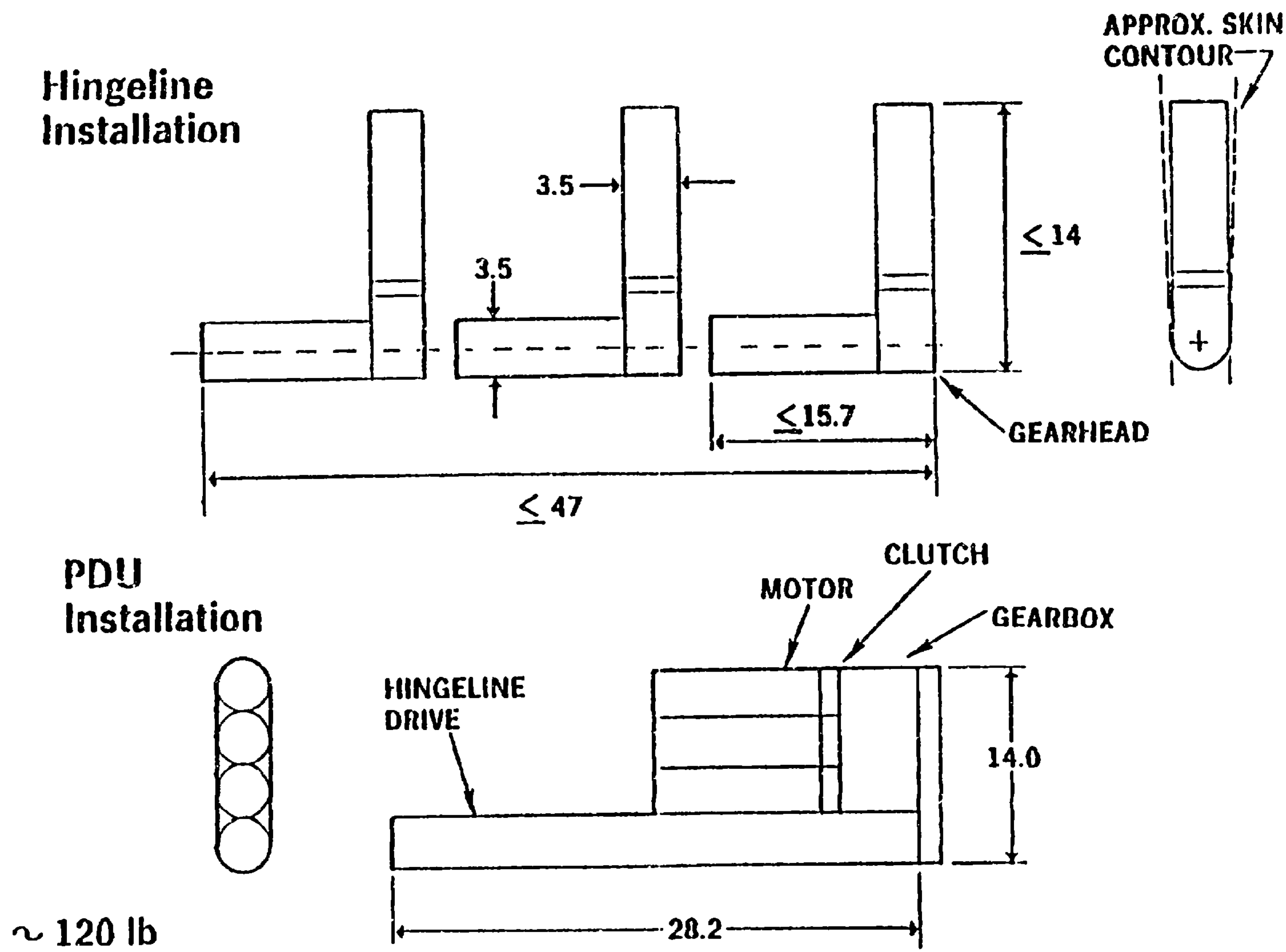


Figure 25. Inboard Flap Installation.

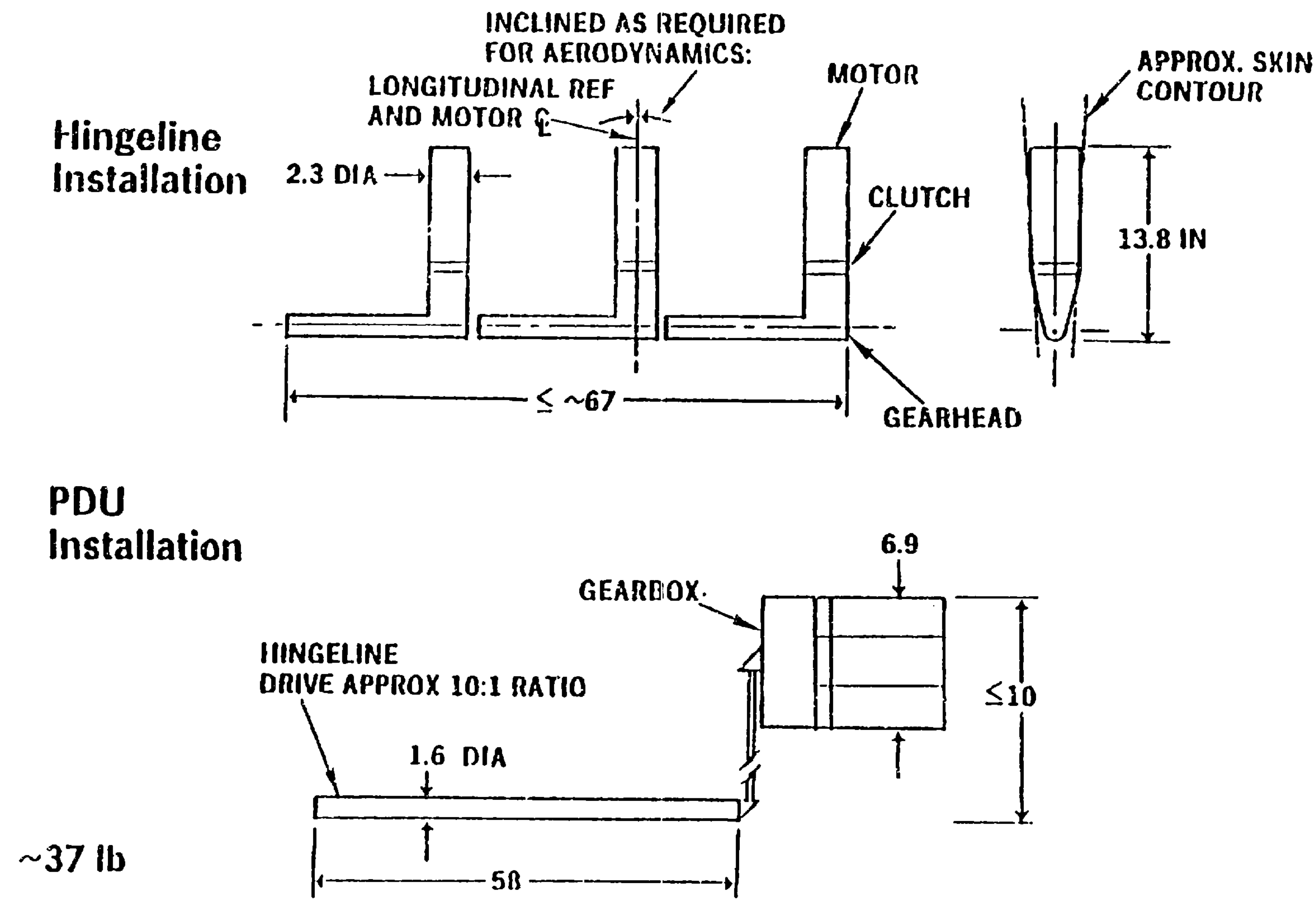


Figure 26. Midspan Flap Installation.

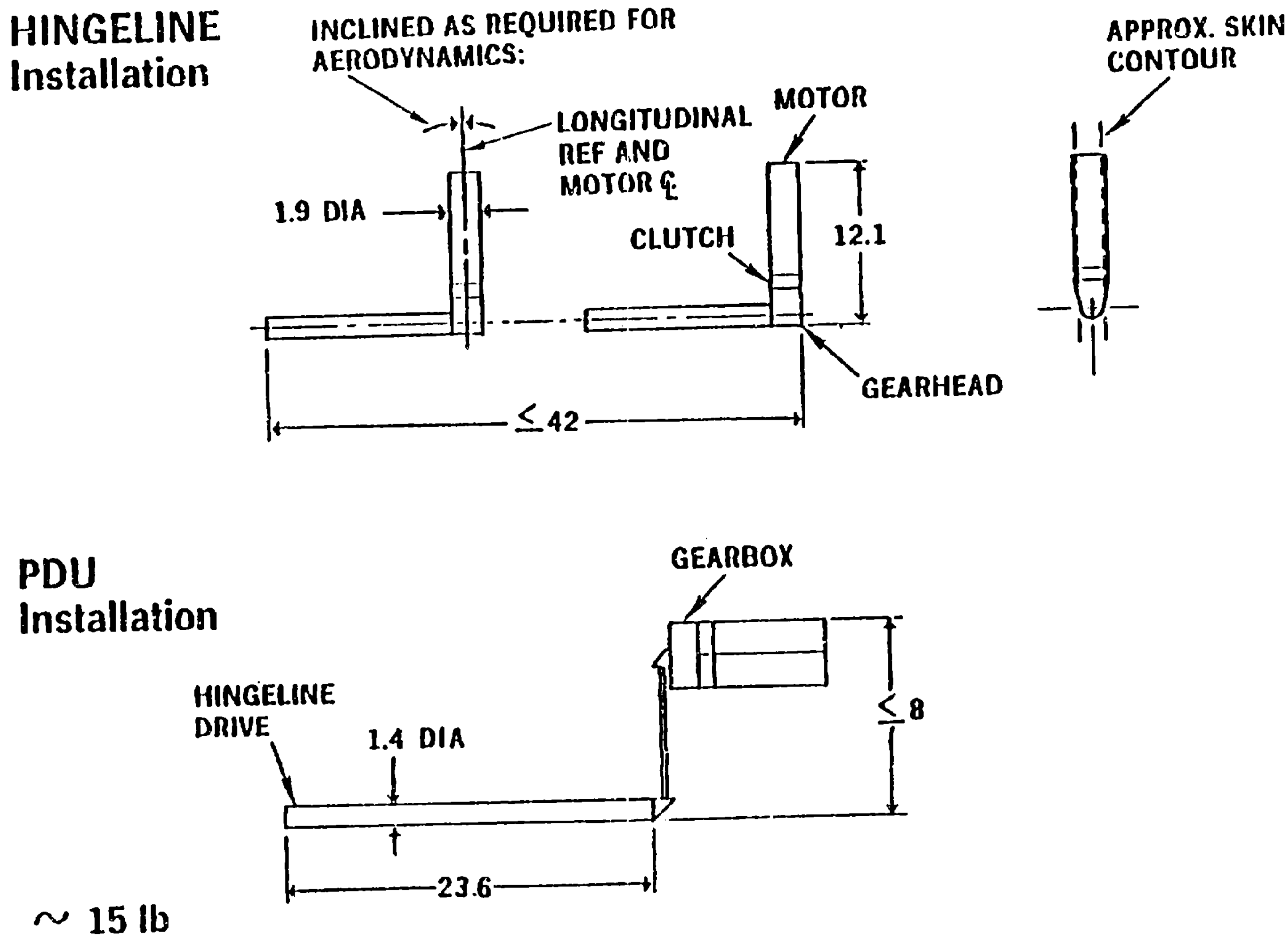


Figure 27. Aileron Installation.

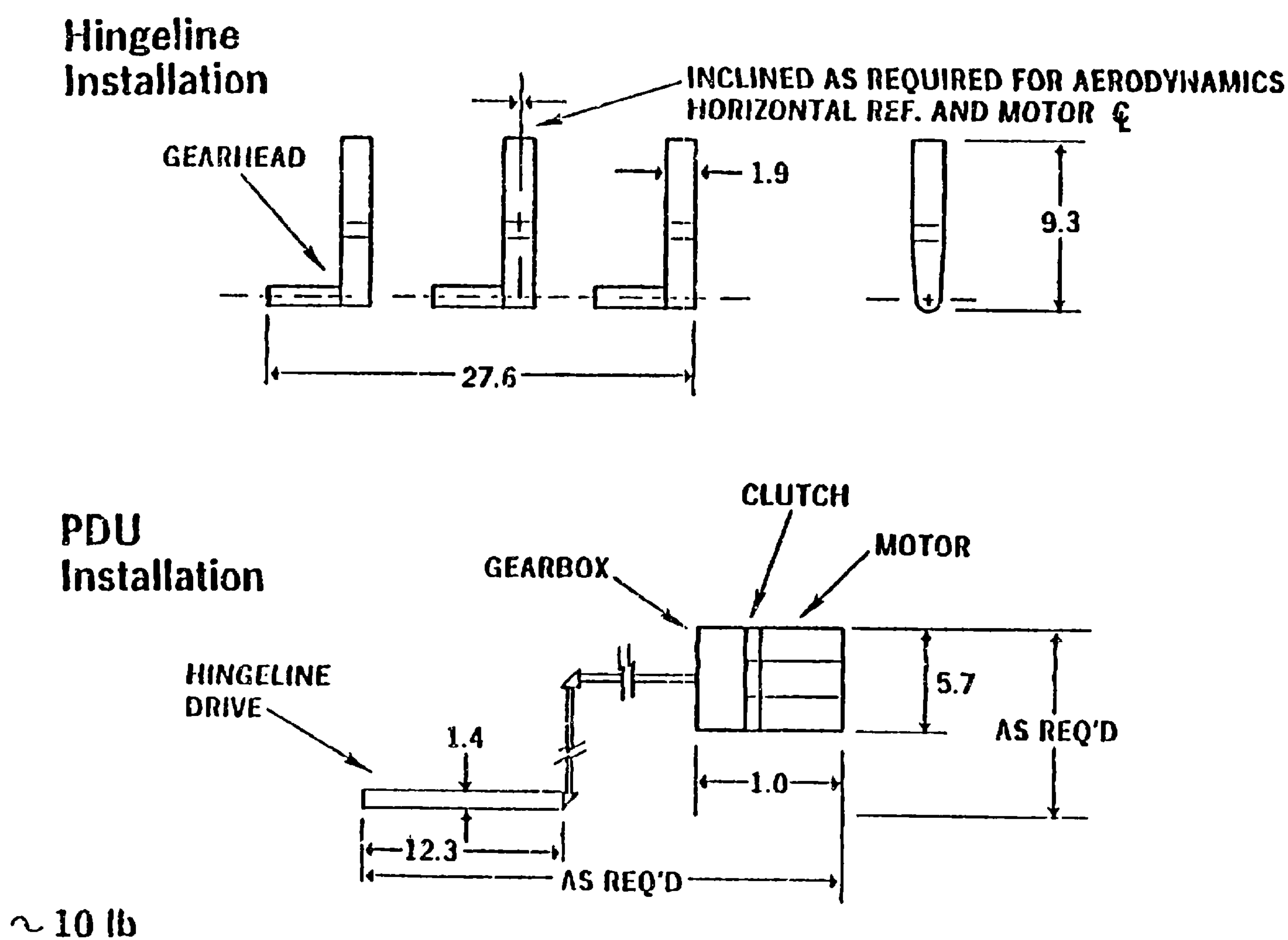


Figure 28. Upper Rudder Installation

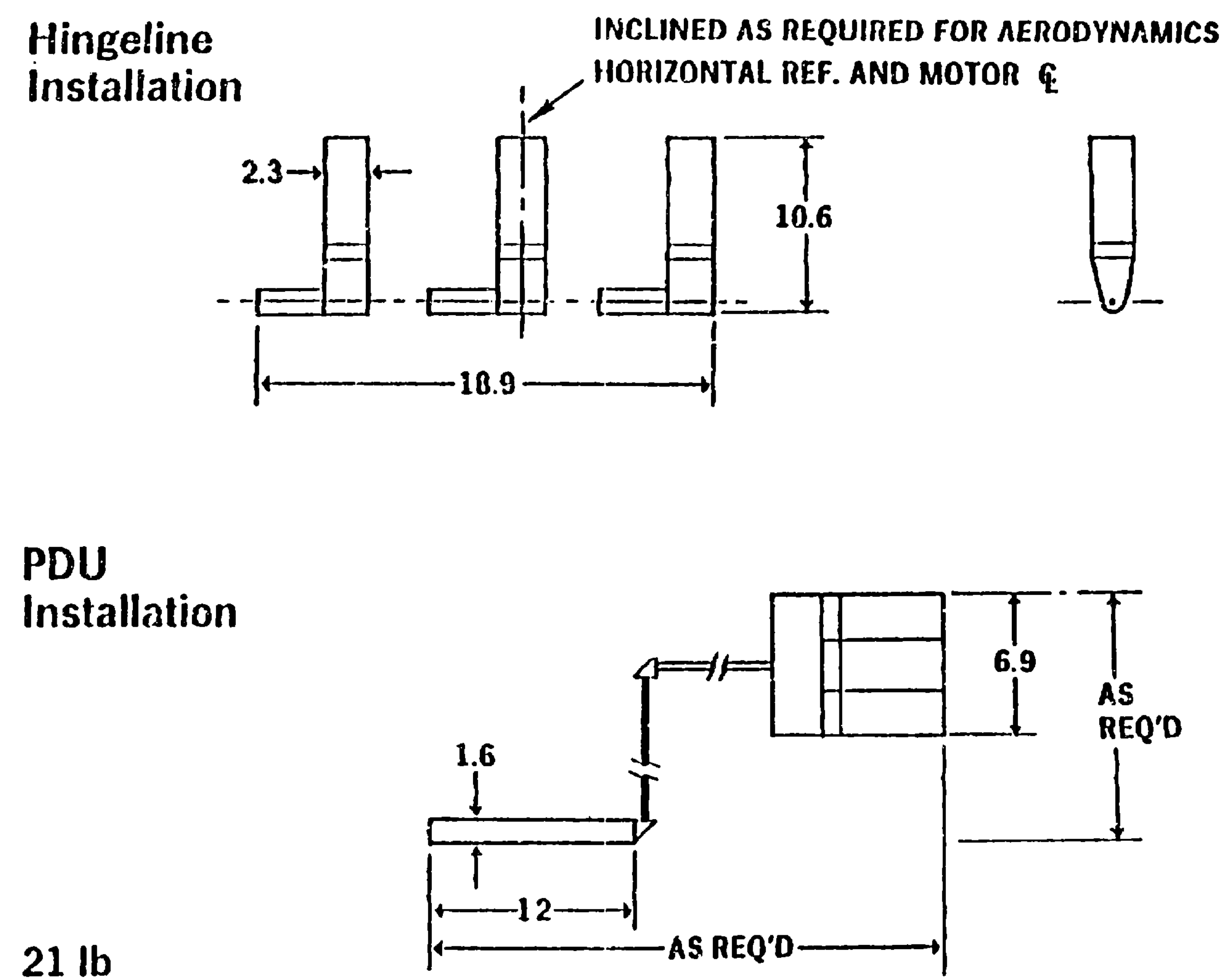
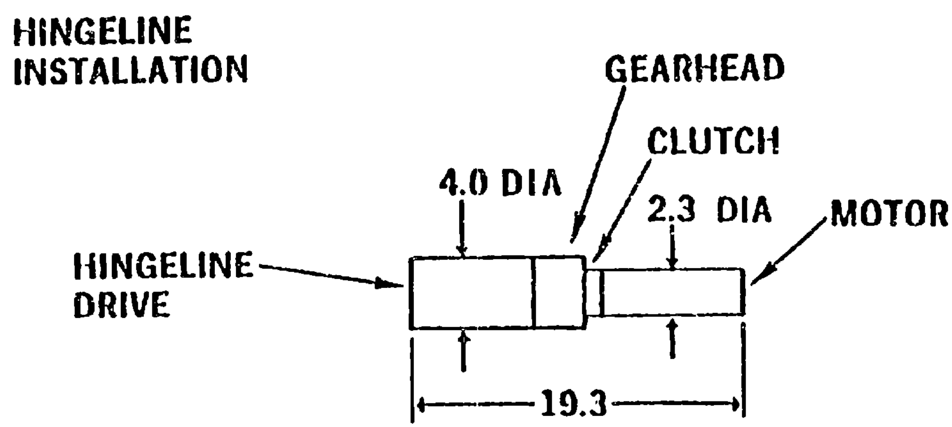


Figure 29. Lower Rudder Installation.



20 lb (~240 lb for 12 actr's)

Figure 30. L.E. Flap Installation

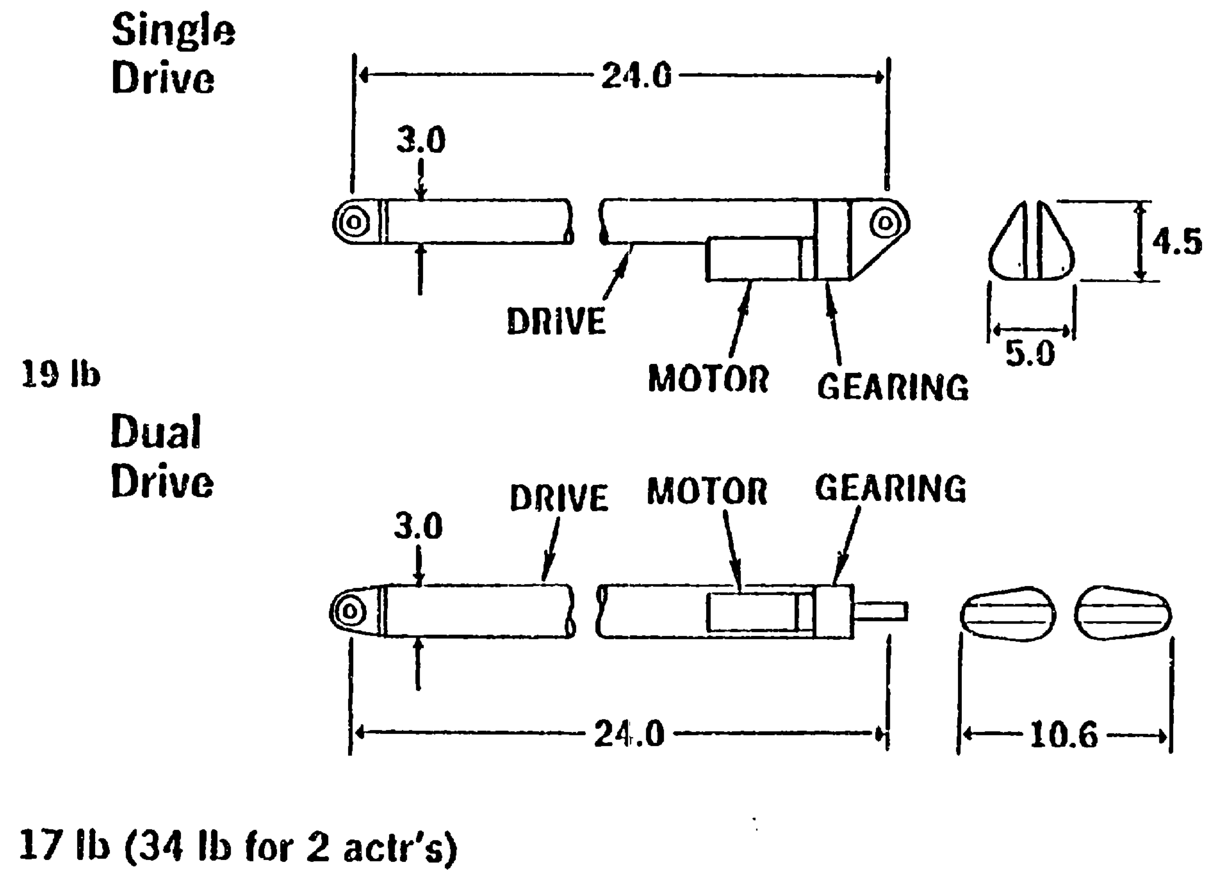


Figure 31. Canard Installation.

It can also be seen that in all actuation applications, exc pt that for the canard, the actuation system was built around hingeline act.ators. In the case of the canard a ball screw was used as the final output loa device. Studies such as those for the B-1 rudder have consistently shown that, where the input is derived from a rotary power device (hydraulic or electrical motors) and the output is to a surface having a long slender hingeline, the best power transducer is a hingeline compound planetary gear type drive (power hinge). This device provides the greatest hinge moment capabilities and highest stiffness at minimum weight.

Two general approaches are shown in Figures 25 through 31. One uses discrete power hinges and electric motors for each power input channel. The other combines the output of the three discrete motors in a single adapter gearbox powering a single longer power hinge. The first approach, called the "hingeline installation" in the illustrations, had the highest potential reliability because a "disconnect" type failure downstream of the clutch (i.e. in the gearhead or in the power hinge) would not cause loss of control of the surface. The second approach, called the "PDU installation" in the illustrations, would fail destructively in the presence of a disconnect. In spite of this fact the "PDU installation" approach was preferred. This arose from the fact that, for equivalent reliability, the "PDU" approach could be smaller, lighter, and more adaptable for installation. As an example, considering the inboard flap, each power hinge and gearhead when used in the "hingeline" approach must be capable of 70% of rated output load (Reference Figure 2-5). This meant that the three power hinges and gearheads attached to each surface must have a total capacity, considered as a unit, of 210%. In other words this approach, though safe, was larger and heavier than it needed to be by a factor approaching 2.1. In contrast the single power hinge used in the "PDU" approach could theoretically be sized to 100% capacity. In actuality, by sizing it at 150%, the reliability of the unit would closely approach 1.0 and would equal or exceed that of the "hingeline" approach. On this basis the relative weights of power hinges and gearheads for the two approaches would be in the ratio of 1.5/2.1 or the "PDU" approach would weigh 28% less than the "hingeline" approach.

The projected weights of the various installations are shown in the lower left hand corner of the illustrations in Figures 25 through 31. In each instance (except for the installation in Figure 31) the weight quoted was that for the "hingeline" installation. The weight for the "PDU" installation was approximately 20% less, and where applicable, was quoted as the second entry on the Figure.

The motor powers listed in Table 20 were consistently higher than the power per surface requirements given in Table 8. This arose from the fact, illustrated in Figure 32, that the motor was current (torque) limited. A motor which would meet the stall torque (θ=0) without overheating had excess power at design load/rate conditions.

One of the most important conclusions drawn from the various installation illustrations shown in Figures 25 through 31 was the fact the "PDU" approach would fit within the installation envelope for the outboard trailing edge (aileron). It had been felt that, because of its very shallow chord, it would be impossible to install an actuation system in this area without using chordwise blisters. If this had been necessary it would have imposed a significant drag/weight penalty.

The design data given in Tables 18, 19, and 20 and in Figure 25 through 31 was developed using the Airesearch T1-59 "RAATS" program. A typical example of the analytical procedures used and the computer printouts developed are shown in Appendix A. The particular example used in Appendix A represented the first cut at sizing the inboard flap actuation system. In this case each of the three motors, gearheads, and hingeline drives making up the complete actuation system were being sized as 100% units (i.e. any one of the three-power trains attached to the surface could meet 100% of the stall hinge moment requirement of the surface) rather than as 70% units, which was the value later established as the basic requirement for this surface (See Table 11).

4.1.5.1.1 <u>Actuation System Detail Design</u> - In order to lend credibility to the weight, performance and envelope projections made in Tables 18, 19 and 20 and Figures 25 through 31, it was decided that a detail design of at least one of the actuation functions should be made. It was felt that, if the resulting design matched the projected weight and envelope of a critical function within a reasonable degree ($\pm$ 5%), it could be expected that the other projections were probably equally accurate. The inboard trailing edge surface (inboard flap) actuation system was selected as the function around which the design would be based. This function was selected because it represented the most severe combination of design requirements, exhibited by any of the flight control actuation functions, in terms of power, load, frequency response and failure mode. The Airesearch report covering this design is included in this report as Appendix D.

As can be seen in the report (page 2, 5, and 6 of Appendix D) the design met envelope and performance requirements. However, in the report a question was raised (comments and recommendations page 5 of Appendix D) regarding the need to design to the force summed stalled torque of three motors. Although it was probable that the design torque for the hingeline gearbox could be reduced to a value, closely approaching the 216 x 10^3 in-lb value mentioned in the discussion, it was not done for the following reasons:

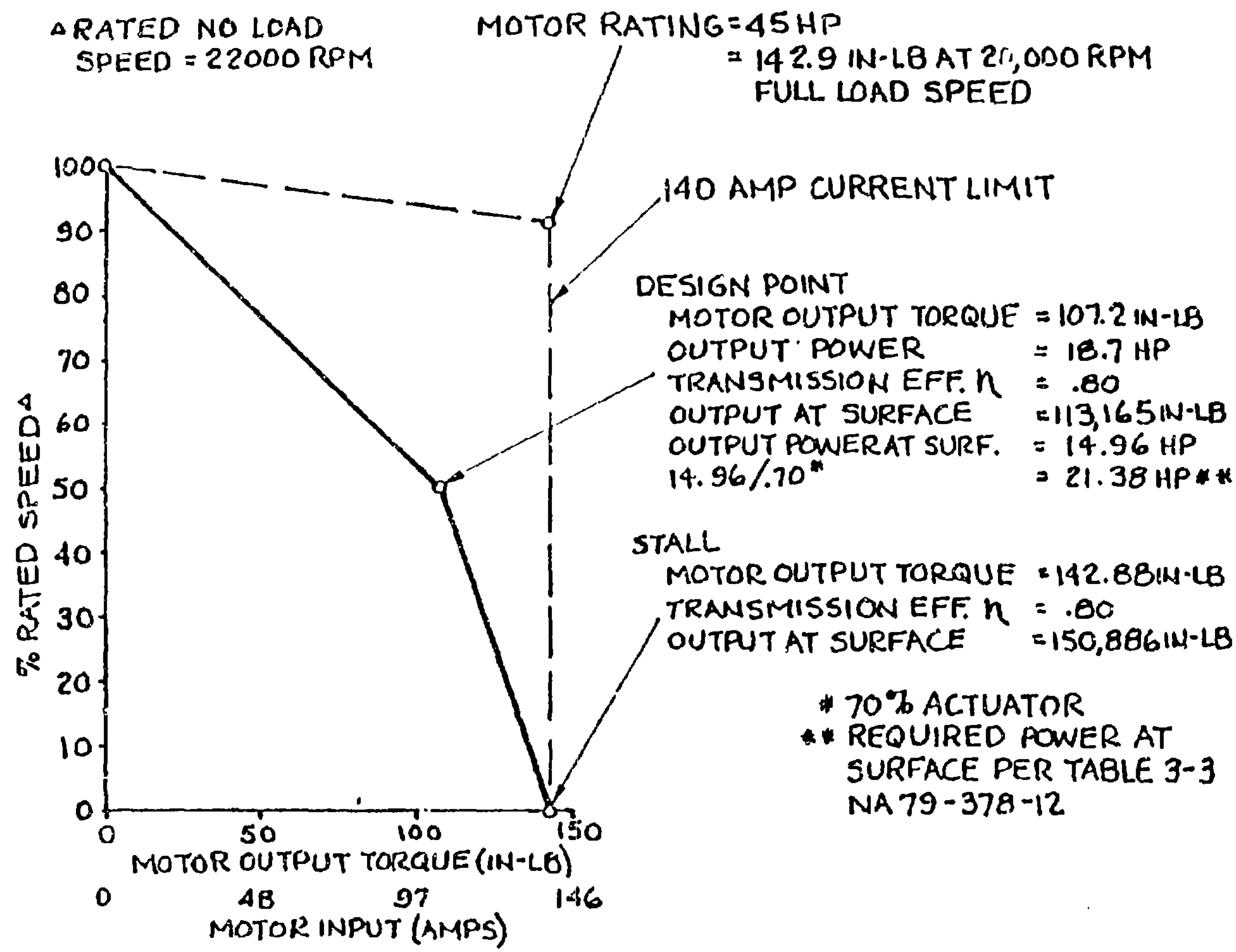


Figure 32. Inboard Flap Motor Characteristics.

1. Since there was no redundancy downstream of the motors (i.e. in the PDU, the reduction gearbox, the two torque tubes or the hingeline gearbox) the downstream items needed a reliability very closely approaching one (at least .9999). To achieve this the design needed to be considerably over designed by normal standards. The over design represented by 453×10^3 in-lb capability (210% of required stall torque) versus an actual maximum applied load of 216×10^3 in-lb (100%) appeared reasonable.

2. Even though, during normal operation, the inboard trailing edge surface would never hit a stop (the electronic controls would determine travel limits) there was always the possibility that, during maintenance operations or during some special in flight failure mode, the surface would inadvertently bottom out. To avoid costly damage the unit should be designed to withstand this condition.

3. Even though it was extra heavy, the gear train and all items down stream of the motor would be identical for Aircraft I and Aircraft II. Therefore, the extra weight did not represent a delta weight for comparison purposes in the study. (Refer to paragraphs 4.2.1.6 and 2.3.5).

The detail design of the inboard flap actuator is shown in Airesearch drawings 2022794, 796, 798, and 824 included as part of Appendix D. Drawing 2022824 represents the overall actuator system and shows that the total weight was 100 lbs. Of this 45 lbs represented the weight of the power drive unit (PDU) shown in drawing 2022798, the electric motors constituted a major share of the PDU weight at 36 lbs (12 lbs each). Page 2 of Appendix D lists the total weight of the inboard trailing edge (flap) actuation system as 220 lbs per surface. Since there were two such surfaces, the total system weight is 440 lbs of which 300 lbs are components subject to replacement by equivalent hydraulic components during the trade study. The weight of these replaceable components was arrived at as follows:

6 inverters at 38 lbs each	=	228 lbs
6 motors at 12 lbs each	=	72 lbs
		300 lbs

The electric motors were rated at 45 HP. This was considerably above the 21.38 HP(15.95 KW) shown in the table of basic flight control actuator requirements Table 8. The reason for this is shown in Figure 32. The figure shows that the motor power capability was determined by the stall torque requirement. To prevent overheating, electric motors of this type are current limited. The current limit required for this application is shown on Figure 32 as 140 AMPS. At this current the motor would generate 142.88 in-lb of torque which, when appearing at the control surface after going through the transmission elements,

was equivalent to 150,886 in-lb. Each of the three motors used for the inboard flap actuation was capable of handling 70% of the required stall torque and hence was commonly thought of as a 70% actuator.

Figure 32 shows that there was a 2000 RPM speed drop in going from no load to full load. At this speed (20,000 RPM), and operating at the current limit, the motor delivered 45 HP. However, the design point load capability requirement was only 18.7 HP at 50% rated speed. This translated to 14.96 HP at the control surface, due to the inefficiencies of the intervening transmission elements, and represented 70% of the 21.38 HP (15.95 KW) required in the basic requirements table (Table 8). It is interesting to note that the motor would very nearly equal the 100% output requirement (19.94 HP versus the required 21.38 HP) at the design point. Thus the motor, although rated as a 70% motor based on stall capability, was actually a 100% motor at operating speeds slightly above the design point. This meant that, even in the face of two failures, the output performance of the actuation system would be essentially unimpaired in most of the practical areas of the flight profile.

4.1.5.1.1.1 <u>Power Drive Unit Design</u> - The power drive unit (PDU) is shown on drawing 2022798 in Appendix D. The drawing showed that the unit was powered by three 270 VDC permanent magnet brushless motors mounted on a torque summing spur gear type gearbox. The motors drove the gearbox through a dog clutch which could be disconnected (but not reconnected) while the motor was rotating and while it was transmitting torque. The dog clutch was normally maintained engaged by a spring and could only be disengaged by energizing a declutching coil (shown on the drawing). Although not shown on the drawing, provisions were made so that the clutch, once disengaged, could only be re-engaged manually with the whole PDU inoperative. This eliminated the possibility that the failure of the coil from overheating while holding the dog clutch disengaged, or from a failure due to wire breakage or other electrical interruption, could allow the dog clutch to attempt to reengage when in motion. This could lead to failure of all three channels in the PDU. The drawing also showed a rotor position sensor, used for commutation, at the anti-drive end of the motor. The maximum output speed of the PDU was 22,000 RPM which was the same as the motor unloaded speed.

4.1.5.1.1.2 <u>Gearbox Design</u> - The gearbox was shown in drawing 2022796 Appendix D. This unit performed the combined function of a right angle gearbox and a speed reducer coaxial with the control hingeline. As shown the speed reduction ratio of the unit was 88:1 via a compound planetary gear train at 91.3% efficiency. By having the first significant gear reduction in this unit, rather than in the PDU, the relatively long shaft between the PDU and the gearbox was small, high speed, low torque, and lightweight. The relatively large, high torque output shaft was short, hence, its weight impact too was minimal.

4.1.5.1.1.3 Power Hinge Design - The power hinge design is shown on drawing 2022794 of Appendix D. It was a classic power hinge design of a type with which Airesearch has had considerable experience. The only unique feature was the relatively large number of "slices" used. A "slice" consisted of all the elements of a power hinge (i.e. two stationary ring gears and mounting lugs, one moving ring gear and its mounting lug, a set of planets with two radial loading rings and a sun gear) and this design used 14 of them. The gear reduction ratio was a relatively modest 15:1 at 88.4% efficiency. The drawing also showed the very high stiffness of the power hinge which was 32 times the stiffness requirement established in Table 8.

4.1.5.1.2 Power-By-Wire/Fly-By-Wire Control System Definition - The basic power control system was defined by Aireseach as a part of their subcontract effort on this program and is included in this report as Appendix B. The discussion in Appendix B showed that the two basic elements of the power control system, for modulated actuators, were the inverter and the controller.

4.1.5.1.2.1 Inverter Description - The functioning of the inverter is described in Appendix B. Essentially the inverter chops and pulse width modulates the 270 VDC power supplied by the electrical power system to cause the actuator's permanent magnet motors to operate bi-directionally at infinitely variable speeds in response to command signals received from the controller. The detail functioning of the inverter is shown on pages 2 through 8 in Appendix B.

4.1.5.1.2.2 Controller Description - The functioning of the controller is also described in Appendix B. Essentially the controller's function was to monitor feedback from the electro-mechanical actuator's output and, using this information, modify and reprocess the flight control system's input signals so that the resulting signals could be used to properly control inverter power to achieve the desired actuator output. The detail functioning of the controller is discussed and illustrated on pages 1 through 3 in Appendix B.

4.1.5.1.2.3 Inverter Design - The detail design of the inverter is outlined in Appendix C. It can be seen in this appendix that the inverters were supplied in three basic sizes to cover the actuator load requirements assigned to Airesearch for their study. Actually a fourth and much larger size was subsequently found to be necessary to meet the requirements of the plug throat. Although the size and weight of the total complement of inverters used in the aircraft could have been reduced by tailoring each inverter to its load application, or at least by increasing the number of sizes, it was felt that 4 sizes represented the optimum compromise between volume/weight versus logistics/maintainability in terms of life cycle costs. The four inverter sizes and some of their critical characteristics are shown in Table 21. This table repeats the data already given in Tables T-1 and T-2 of Appendix C and uses the data for the 3 original inverter types shown therein as a basis for extrapolating the fourth inverter type (i.e. the plug throat inverter).

TABLE 21. INVERTER CHARACTERISTICS

CHARACTERISTIC	APPLICATION			
	PLUG THROAT	INBOARD FLAP	EXTERNAL FLAP MIDSPAN FLAP	THRUST VECTOR VANE AILERON CANARD LEADING EDGE FLAP UPPER/LOWER RUDDER
CURRENT RATING	350 AMP	150 AMP	50 AMP	25 AMP
POWER RATING	94.5 KW	40.5 KW	13.5 KW	6.75 KW
DIMENSIONS				
DIAMETER	9.0 IN	7.2 IN	5.0 IN	
WIDTH				7.0 IN
DEPTH				3.0 IN
LENGTH	23.6 IN	18.7 IN	10.4 IN	11.0 IN
VOLUME	1500.0 IN^3	761.4 IN^3	204.2 IN^3	231.0 IN^3
WEIGHT	67.5 LB	38.0 LB	12.5 LB	10.0 LB
COOLING*	EC	EC	EC	NC

* NC = NATURAL RADIATION/CONVECTION COOLING
EC = EVAPORATIVE COOLING - FINNED OUTER SURFACE WITH FORCED CONVECTION, 130°F, 30 CFM/FT OF LENGTH, 2 IN H_2O ΔP, OR IMMERSION IN FLUID HEAT SINK

4.1.5.1.2.4 Inverter Cooling - The 54 inverters used in aircraft I will, as a group, typically reject 4.03 KW in the form of heat during longest high output sustained duty cycle (terrain following - 32 Min). Heat rejection of this magnitude was felt to be a potentially serious problem especially when considering the fact that a large proportion of the heat would be rejected from very small components (i.e. the field effects transistors "FETs" used in the inverters). For this reason both Rockwell and Airesearch studied the problem and came to generally the same conclusions. These conclusions were that inverters rated at less than 25 amps could be cooled by natural conduction and convection and that inverters rated at 50 amps and above must be evaporatively cooled. Airesearch's analysis of the subject is contained in Appendices D-1, D-2 and D-3 to Appendix C. Rockwell's analysis is discussed in the following paragraphs.

Ideally the electro-mechanical actuator should reject its internally generated heat (i.e. that resulting from motor or control system inefficiencies) to its immediate surroundings. By so doing, a system using electro-mechanical actuators could avoid the need for auxiliary cooling ducting or numerous liquid cooling lines spreading out through the aircraft to service each actuator. If such a spiderweb of lines and/or ducts were to prove necessary, it was felt it would offset a large portion of the advantages derived from deleting the hydraulic system. Tests conducted and reported in reference 17 show quite conclusively that the electric motor/power hinge portion of an electro-mechanical actuation system can reject it's self generated heat to its immediate surroundings. However the analysis made in Appendix C showed that auxiliary cooling aids were required where large sized inverters were a part of the actuation system.

Table 22 shows the heat rejection characteristics of the inverters for all the various flight control (continuous duty type) actuators and two engine actuation functions. These two engine actuation functions were included because, even though classified as utility functions, they were modulated and had continuous duty characteristics. The other utility actuators were not included in the table because, in general, heat rejection was not a problem for this type of actuator. In most instances this was because they are not continuous duty, and therefore their operations were infrequent, and their operating times were short. Continuous duty elements in the environmental control system, such as pumps, and blowers, were also not included because they did not employ inverters and because they were an integral part of, and could reject such heat as they did generate to the ECS system. This, plus the fact pointed out earlier, that the ECS system for all study aircraft would be essentially the same, justified their elimination. Table 22 is an expansion of the data given in the table on page D-1-1 of Appendix C and the new additions were an extrapolation from the data on which that table is based.

TABLE 22. INVERTER LOSSES FOR 25% DUTY CYCLE*

ACTUATOR	MOTOR CURRENT (AMPS)	INVERTER TYPE **	INVERTER RATING (AMPS)	25% CURRENT (AMPS)	LOSSES PER INVERTER (WATTS)	TOTAL NO. OF INVERTERS	TOTAL LOSSES (WATTS)
INBOARD FLAP	156	EC	150	39.0	316.9	6	1901
MIDSPAN FLAP	31	EC	50	7.8	56.9	6	341
AILERON	14	NC	25	3.5	6.6	4	26
UPPER RUDDER	3.5	NC	25	0.9	1.6	6	10
LOWER RUDDER	6.9	NC	25	1.7	3.0	6	18
L. E. FLAP	17	NC	25	4.3	8.7	12	104
CANARD	3.5	NC	25	0.9	1.5	4	6
THRUST VECTOR VANE	10.2	NC	25	2.6	4.7	4	19
EXTERNAL FLAP	59	EC	50	14.8	113.5	4	454
PLUG THROAT	364.6	EC	350	91.1	701.0	2	1402
						Σ LOSSES	4281

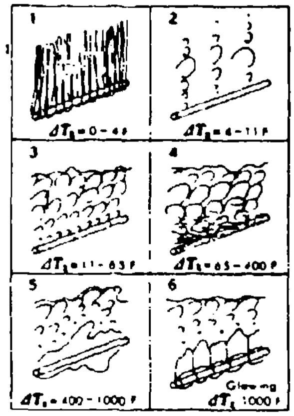
* DATA FOR 50% SPEED
** COOLING METHOD – EC = EVAPORATIVE COOLING
– NC = NATURAL (CONDUCTIVE/CONVECTIVE) COOLING

Of the actuation functions listed in Table 22 only the first and last two rejected significant amounts of heat. For the other types of actuators the heat rejection was low enough so that all the heat generated could be rejected to ambient air directly or through the actuator's mounting pads and thence via structure to ambient air.

The high heat rejection functions involved the 18 actuators used for powering the inboard and midspan flaps on the left and right side and the two engine functions (plug throat and external flap). These actuators required a more exotic cooling method. The method which appeared best was some form of evaporative cooling based on the nucleate boiling of a dense, inert, low viscosity fluid.

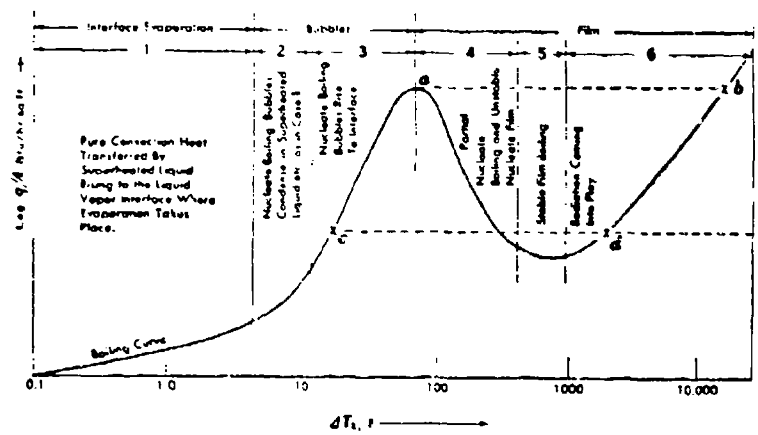
Figure 33 illustrates the mechanism of nucleate boiling. It shows that accomplishing component cooling in phases 1, 2 and 3 (Figure 33) was the most effective and indicated that the component surface temperature should never be more than the peak value, shown at "a", above the coolant's temperature. The Figure 33 data showed, in both the main figure and in block 3 in the pictorial illustrations of boiling in the upper part of the figure, that the peak ΔT was 65°F for water. The equivalent value for a typical inert cooling fluid, such as freon 113, was approximately 35°F. The difference resulted largely from the cooling fluid's specific heat. Evaporative cooling for electronic components involved a regenerative cycle which consisted of nucleate boiling followed by vapor condensation in a closed system. Figure 34 illustrates this general approach. It shows three circuit boards, mounting high output electronic devices, installed in a sealed housing and immersed in a coolant fluid. The electronic devices were rejecting sufficient heat so that the fluid was boiling at a relatively high rate (phase 3 in Figure 33) which was sufficient to cause the vapor bubbles to rise through the liquid and escape to the vapor zone. In the meantime cooling airflow, or a heatsink fluid, was circulated over the finned outer surface of the housing and cooling it sufficiently to cause the vapor to recondense. When cooling airflow was used it was induced by convection when the aircraft was on the ground and by ram effects or forced cooling air in flight.

A reasonably well designed forced convection air cooling system would remove 0.05 watts of rejected heat for each square inch of cooling surface per degree centigrade differential temperature (0.05 watts/in^2/°C). A comparable figure for an evaporative cooling fluid was 1.5 watts/in^2/°C). In other words cooling fluids were 30 times as effective at removing heat from a surface as was air. It was, therefore, reasonable to assume that, when air cooled, the area of the outer finned surfaces of a unit, such as Figure 34, should be at least 30 times the effective heat transfer area of the heat generating electronic componenets themselves. This assumed that ΔT between the electronic component and the fluid approximately equaled the ΔT between the vapor and the cooling air.



NOTE:

1) ΔT_x = Temperature of heating surface minus the liquid boiling temperature.



Typical boiling data for a wire heated electrically in a pool of water at atmospheric pressure. (Extracted from "Heat Transfer to Water Boiling Under Pressure," by E. A. Farber and R. L. Scorah, published in *Trans. ASME*, Vol. 70, 1948, with permission of the publishers, The American Society of Mechanical Engineers)

1 - Free convection heating (no boiling).
2 - Nucleate boiling - Bubbles condense in liquid.
3 - Nucleate boiling - Bubbles rise through liquid and escape into vapor zone.
4 - Partial film boiling - Bubbles are formed so fast on the heating surface that part of the heating surface is covered with a vapor film. This vapor film insulates the heating surface, decreasing the heat flux.
5 - Film boiling - The heating surface is completely covered with a vapor film.
6 - Radiation - Radiation heat transfer dominates the film boiling.

Figure 33. Heat Transfer with Change in Phase.

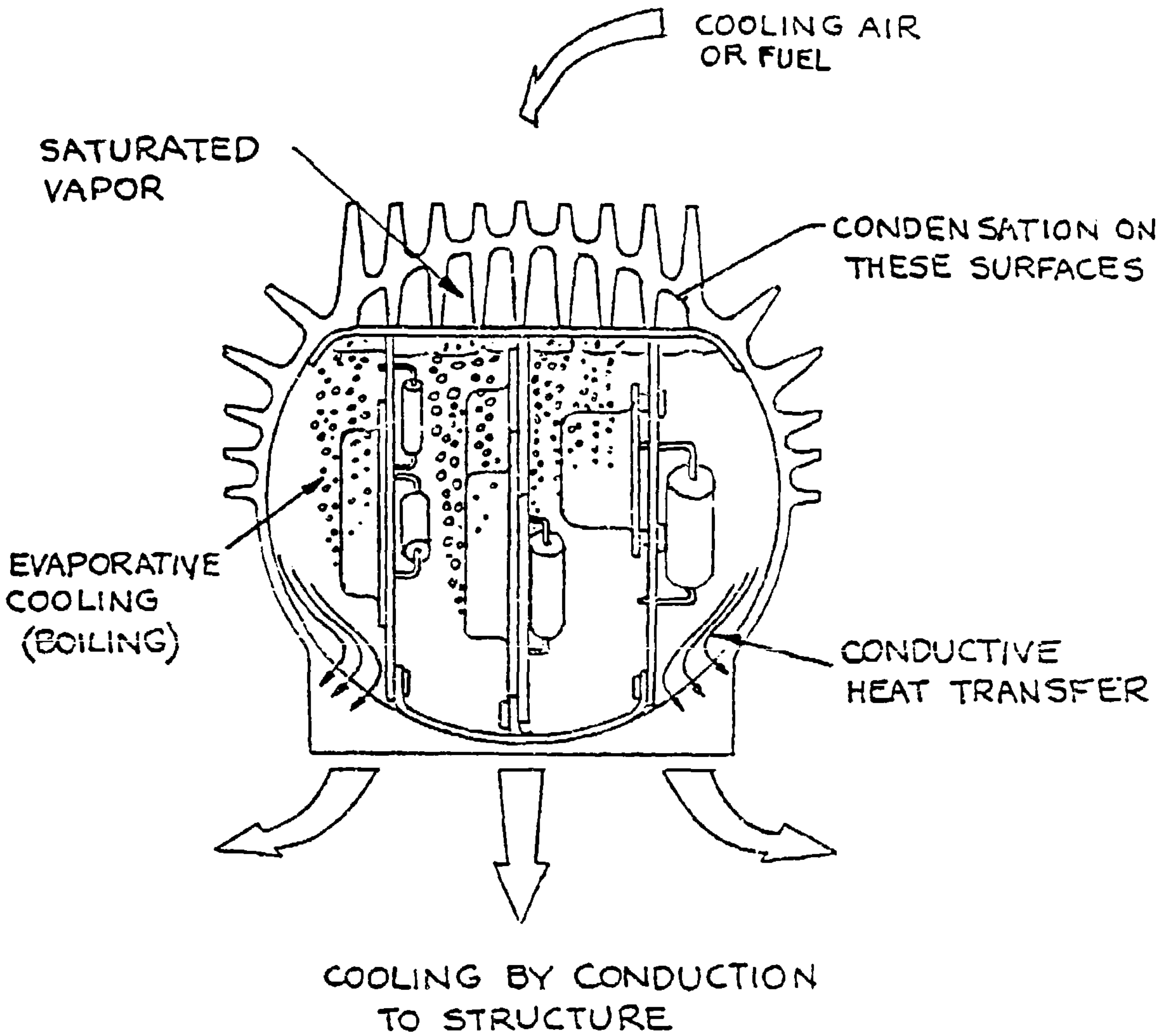


Figure 34. Evaporative Cooling of Electronic Equipment.

TABLE 23. EVAPORATIVE COOLING FLUID PROPERTIES

CHARACTERISTIC	CANDIDATE FLUID		
	FREON 113 ①	FC 78 ②	FC 75 ②
SPECIFIC HEAT @ 73°F (C_p) (BTU/LB/°F)	.214	.240	.245
HEAT OF VAPORIZATION @ 73°F (h_{fg}) (BTU/LB)	66	41	38
SPECIFIC GRAVITY @ 73°F (G)	1.57	1.70	1.76
DENSITY @ 73°F (γ) (LB/FT^3)	98	106	110
VISCOSITY @ 73°F (μ) (CENTIPOISE)	.64	.40	.81
BOILING POINT @ 1 ATM (°F)	130	122	216
VAPOR PRESSURE @ 73°F (P_g) (mm Hg)	310	260	35
THERMAL CONDUCTIVITY (BTU/HR/FT^2/°F/FT)	.043	.036	.037
COEFFICIENT OF EXPANSION (FT^3/FT^3 °F)	NO DATA	.0009	.0009
POUR POINT (°F)	-31	-135	-135
MATERIAL COMPATIBILITY	GOOD	EXCELLENT	EXCELLENT

① DU PONT DE NEMOURS & CO.
② MINNESOTA MINING & MFG. CO.

Table 23 lists the basic characteristics of three potential evaporative cooling fluids. Of the three, Freon 113 appeared to offer the best balance of properties. It had the highest heat of vaporization (i.e. about 40% greater than that of the other two fluids, but still only 7% of that of water), and the highest thermal conductivity. It was also the lowest in weight (i.e. roughly 10% less than the other two fluids but still nearly 60% heavier than water). The only areas where Freon 113 appeared to be deficient, relative to the other two fluids, was in pour point and material compatibility. It was desirable, but not mandatory, that the fluid have a pour point below the minimum operating temperature (-65°F). However, in a passive system, such as that shown in Figure 34 where fluid circulation is not required at low temperatures, the only adverse impact of a high pour point is the possibility, that at lower operating temperatures, the fluid will freeze in a damaging manner deforming encapsulated components. It is not believed that, at -65°F, Freon 113 would freeze solid enough to damage components.

Freon 113 was also somewhat deficient, with respect to the other two fluids, as regards its long term inertness relative to some materials of construction (specifically silicone compounds) commonly used in electronic hardware. Freon contained both flourine and chlorine in its molecular structure. Chlorine would tend to attack some silicone compounds as well as some highly stressed metals under certain conditions. The "FC" fluids contained only flourine as the halogen in its molecular makeup. For this reason the "FC" fluids were almost perfectly inert to all materials of construction.

The transistors, used in all continuous duty cycle (flight control type) inverters, had a maximum continuous junction temperature limit of 125°C (256°F) and a short time maximum junction temperature limit of 150°C (302°F). Cooling these transistors would be no problem except under mach 2.2 flight conditions where the cooling air temperature (ram ambient air) was 117°C (242°F). Since this condition could exist for as long as 22 minutes continuously on a single flight (see Figure 1 and Table 3), thermal lag could not be counted on. The maximum continuous transistor junction temperature limit was only 8°C (14°F) above the cooling air temperature. This was an impractically small differential temperature for achieving any significant heat transfer.

There were three possible solutions to this problem.

1. Use higher boiling temperature fluid, such as FC-75 or FC-43, in an evacuated sealed housing.

2. Provide cooling air from the ECS system to the inverters during high speed flight.

3. Immerse the inverters in fuel.

The use of a higher boiling temperature fluid in an evacuated housing would make it possible to maintain relatively low boiling temperatures and hence, relatively low junction temperatures when the available ambient cooling air temperature was low. It also allowed the junction temperature to follow the ambient cooling air temperature, at a relatively constant ΔT, up to the maximum temperature encountered. This is illustrated in Table 24 which shows that, with cruise at 32,000 ft, the ram ambient cooling air was -6°F and the corresponding transistor junction temperature was a chilly +39°F. At the other end of the spectrum the available cooling air reached a maximum of 242°F during mach 2.2 cruise and the corresponding junction temperature became 298°F. This was 4°F under the short time maximum limit of 150°C (302°F). Although the junction temperatures shown in Table 24 were not necessarily accurate they were in the right ballpark and were probably accurate within $\pm 2\%$ for a typical transistor which rejects approximately 12 watts for each square inch of transistor outer housing surface.

This approach (using a sealed housing and allowing the junction temperature to follow the cooling air temperature) largely avoided the problems associated with using a fluid in a constant pressure (and hence constant boiling temperature) housing. If a low temperature boiling fluid had been used, in the constant pressure approach the fluid would have tended to turn completely into a large and unmanageable volume of gas during the high temperature (mach 2.2) portion of this mission.

In affect the cooling would thus have been occurring in the stable film boiling or radiation cooling range shown as zones 5 and 6 in Figure 33. Under these conditions the junction would have been in the 1000°F+ range and would have immediately failed. If a high temperature boiling fluid (262°F at 1 ATM) were used, one which would still be in the nucleate boiling range (zone 3 Figure 33) at the high cooling air temperatures (242° F) associated with mach 2.2 flight, the transistor junction temperatures would have tended to be around or above the allowable maximum continuous junction temperature (256°F) for a greater portion of the transistor's operational service life. Since transistor life was an inverse exponental function of junction temperature this could have had a serious adverse impact on life and reliability.

As an example of conditions existing under other circumstances the junction temperature of a constant pressure boiling system would have been about 272°F during low level terrain following whereas that for a sealed variable pressure approach under the same circumstances would have been as indicated in Table 24. (i.e. 198° F). On an average the transistor junction temperature for the sealed variable pressure approach would have been at least 80°F less than that for the constant pressure boiling system under the operating conditions and flight times logged by the ATS aircraft.

Based on the preceding discussion, the solution offered by item 1 above might have been marginally satisfactory, particularly if a 10 to 20 degree further

TABLE 24. EVAPORATIVE COOLING PRESS/TEMP CONDITIONS

PARAMETER (2)	FLIGHT SEGMENT OR OPERATION			
	CRUISE 32,000 FT. 0.9 M	LOW LEVEL TERRAIN FOLLOWING	SUPERSONIC CRUISE 2.2 M	FILLING HOUSING DURING FAB
JUNCTION TEMP.	+39°F	198°F	298°F	241°F (1)
BOILING TEMP.	+14°F	163°F	262°F	206°F
BOILING PRESS.	.06 PSIA	6 PSIA	38.7 PSIA	14.7 PSIA
CONDENSATION TEMP.	+4°F	153°F	252°F	196°F (1)
COOLING AIR TEMP.	-6°F	143°F	242°F	186°F (1)

(1) A POINT CONDITION PASSED THRU ON THE WAY TO AND FROM SUPERSONIC CRUISE.

(2) ASSUMES USE OF MINNESOTA MINING & MFG. CO. COOLING FLUID FC-75.

increase in allowable transistor junction temperature could have been assumed for the 1990+ time frame. However, even if this approach had been usable, the weight represented by the ducting and equipment necessary to direct the proper amount of cooling air to all 18 of the inverters requiring ram air cooling plus the ram drag rise associated with extracting air from the airstream indicated that it was not one of the better choices.

As indicated in item 2 above, the second approach to cooling the actuators would have been to provide ECS cooling. In this approach the same basic evaporative cooling techniques would still be used. However, in this instance, a low temperature boiling fluid at constant pressure would have been the cooling medium. Evaporative cooling would have been used in preference to liquid cold plate cooling or direct air cooling because evaporative cooling was so much more effective at extracting the heat from the hot spot and transporting it for dissipation over a large surface area. The large surface area would then have been cooled by the ECS system either by air or liquid (coolanol). However, this approach was subject to the same general objections as those cited for item 1 above. This approach would still require either, complex ducting if air cooled, or complex piping if coolanol cooled.

In either event the ultimate heat sink would have been the aircraft's fuel (see Figure 10). This fact lead to the conclusion that the solution offered by item 3 above was the best approach.

Item 3 envisioned immersing the inverters in the fuel tank. This appeared to be a reasonable approach based on the following:

1. The inverters should be close to the generators supplying them (3 to 6 ft).

2. The generators were surrounded by sump tanks (see tanks #3 and #4 in Figure 22).

3. The inverters could be installed in the sump tanks and be within 6 ft of the generators.

4. Rejection of heat to fluid was approximately 30 times as effective as rejecting to air, therefore, by using fuel as a heatsink, the inverter housing could be of minimum size and weight.

5. The aircraft was equipped with an air to fuel heat exchanger in the fuel recirculation loop for alert status ground cooling. This heat exchanger, and its ground cooling fuel heat sink door (see Table 9), could be used for subsonic inflight cooling of the fuel before and after the M 2.2 portions of the mission to ensure a low sump tank fuel temperature (<70°F).

6. The limit temperature for the fuel was 150°F and the sump tank fuel capacities were 2500 lbs. Under these conditions the sump tanks had a heat absorption capacity of 100,000 BTU (2500 lb x 0.5 $^{BTU}/_{LB/°F}$ for fuel x 80°F = 100,000 BTU), in going from 70°F to 150°F during M 2.2 operations.

7. The heat rejected to the fuel during the 22 minutes maximum of mach 2.2 operations was 5128 BTU, which was only 5.1% of the 100,000 BTU capacity, and indicated that, even with the other heat sink demands placed on the fuel, of which the 74,407 BTU placed on it by the ECS system (see Reference 12, page 3-20) is the major item, the heat sink capacity is adequate.

The heat rejection value used in item 7 above was derived from Table 22. This table listed the losses of all the inverters used in the aircraft when operated on a 25% duty cycle. This duty cycle was felt to be representative of the mean loads which would be encountered over the 22 minutes mach 2.2 operation encountered during penetration and combat (see load analysis Table 16, sheet 3). The table included the losses for both the air cooled (NC) inverters and the fuel cooled (EC) inverters. 4098 watts represented the losses for the fuel cooled inverters, out of a grand total of 4281 watts for all inverters, and was the value used to determine the 5128 BTU heat rejection figure used in item 7 above.

4.1.5.1.2.5 Power-By-Wire/Fly-By-Wire Control System Arrangement - Figure 35 represents the general arragnement used for power control in Aircraft I. The arrangement attempted to take maximum advantage of the four indepent-dent electrical systems by combining them with triple redundant actuators and five channel flight control inputs (4 control channels plus a model channel) to obtain maximum reliability. The use of five channels in the "Fly-By-Wire" flight control inputs made possible voting in the face of a third failure. This practically eliminated the possibility of a "hard over" third failure and simplified the achievement of a "fail safe" condition after the third failure. The micro processors in the various systems (system #1 through #4) exchange data and voting information via optical interties. In this way absolute separation of the four power systems was maintained in that there were no electrical interconnections either for power transfer or signal interchange. Optical interties were also used exclusively between the various microprocessors and their respective inverters, motors, and actu-ators as well as for the fly-by-wire (fly-by-light) flight control inputs. Through the use of this approach the potential adverse impact of electro magnetic interference (EMI) generated in the inverters and elsewhere was minimized. The use of optical interties, in the manner indicated in Figure 35, also gave the power control system a high degree of resistance to electro magnetic pluse (EMP) effects such as would be associated with lightening strikes or nuclear blasts.

Figure 35 is an expansion and elaboration of the basic Airesearch block diagrams shown as Figure C-4, page 291, Figure C-5 page 293, and Table C-4 page 294, of Appendix C. In the event of failure of a given inverter or actuator motor in a given system failure was detected in the system's micro processor based upon data fed back from the defective inverter or motor. This information was then sent electro-optically to the appropriate neighboring microprocessor where a signal was generated to cause the appro-priate solid state relay to energize and disconnect the motor clutch in the malfunctioning motor. As a specific example (see Figure 35), if the failure were in the inverter (INV) for the right hand (R.H.) inboard trailing edge (ITE) surface for system #1, the system #1 microprocessor would sense the failure via electro optical feed back from the inverter and send the failure intelligence to system #2 and system #4 microprocessors. These microprocessors would process the information and send a signal to their respective solid state relay banks which would cause the appropriate relay in each bank to energize its declutching coil in the motor for ITE INV R.H. #1. As can be seen in the power drive unit drawing (Airesearch drawings No. 2022798 - Appendix D) the dog clutch must be electrically energized to disengage. Although not clearly shown in drawing No. 2022798, the clutch actuating coil was actually a dual coil powered, in this case, by system nos 2 and 4. Either system by itself was capable of declutching the unit.

The system could have been designed so that loss of power could have caused it to declutch (i.e., spring loaded to declutch). However, this approach

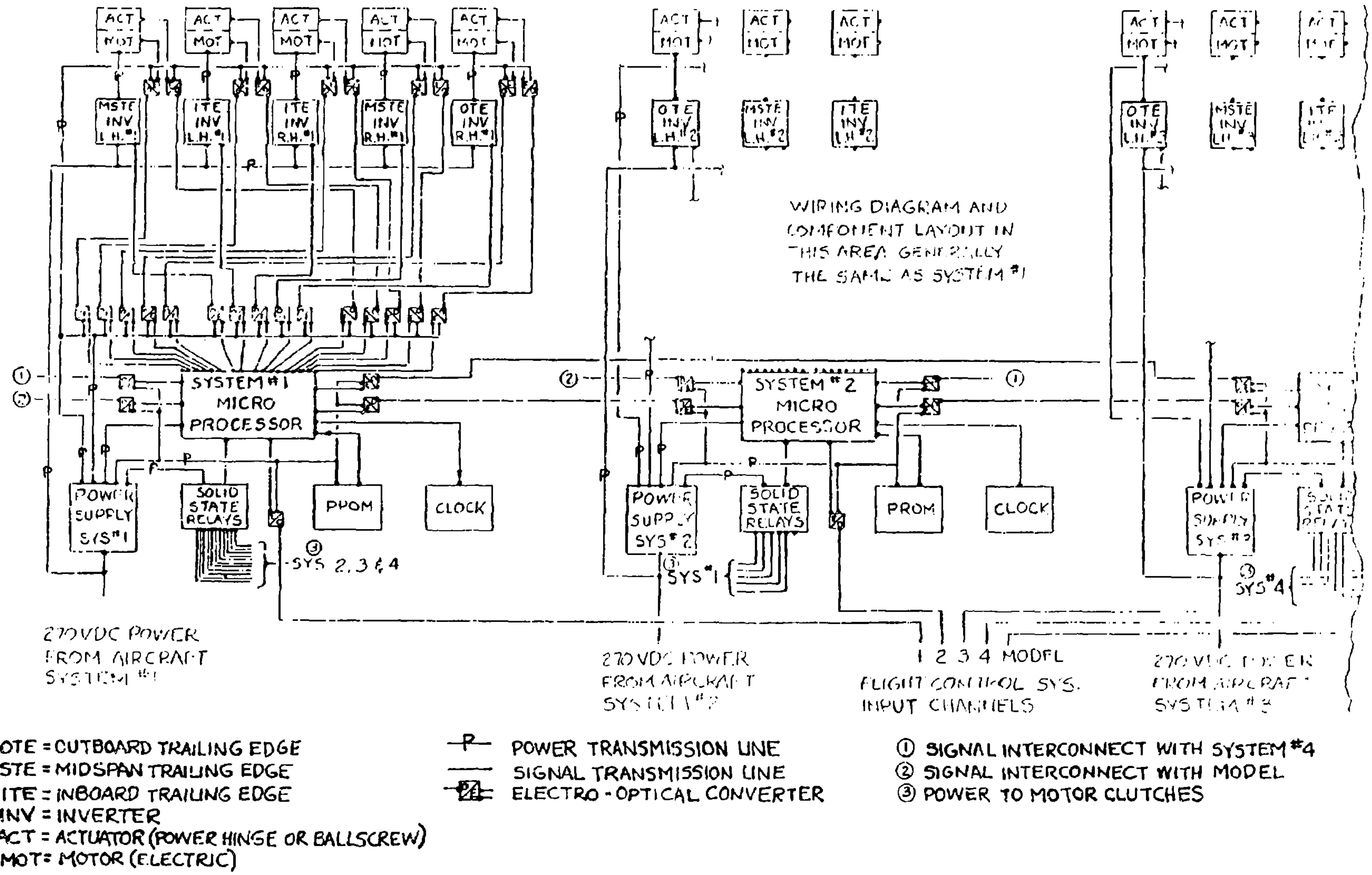


Figure 35. Wing Trailing Edge Surface Power Control

was discarded because it would have meant that all the coils in all actuators would have to be energized at all times during normal operation. This appeared to be an unacceptable heat load and power drain. In contrast, considering the selected design, only those motors which have failed or are part of a system which has failed were energized. In the event of a dual system failure this might have meant that as many as 8 motors (out of 16 in the trailing edge flap system) might have had energized clutches. However, this would have occurred relatively infrequently and since, with the selected design, the coil was never energized while the motor was running the motor heat load was not additive to the coil heat load. In contrast, with the spring loaded to declutch design, the two heat loads would have been continuously additive and would have created a major cooling problem.

As implied above a complete system power failure (generator out) or microprocessor malfunction in system #1 would have caused system nos. 2 and 4 microprocessors to cause all clutches on all motors in system #1 to declutch. The reason for two declutching coils was to meet the two fail operate requirement. If system #1's failure had been preceeded by a failure in the #2 power system, power from system #4 would have been necessary to accomplish the required declutching.

It should be noted, when examining Figure 35, that an electro optical converter was not shown at the inverter end of the microprocessor/inverter signal transmission line. In the interest of avoiding further complexity in Figure 35 it was assumed that this electro optical converter along with its power supply was built into the inverter.

Figure 36 shows the declutching connections for the wing trailing edge actuation system. The reason for the unbalance in the number of connections (i.e. 11 in systems #1 and #4, and 5 in systems #2 and #3, versus an ideal of 8 in each system) was as a result of the unbalanced distribution of flight control actuation functions needed to adapt dual and triple channel actuators to a four channel power system. In effect, as can be seen by examining the electrical system load summation Table 17 the distribution of actuation functions between power systems succeeded in balancing power demand to 40.22 ± 9.22 KW continuous and 63.27 ± 14.87 KW 5 sec loads during combat.

Although the schematic of Figure 35 covered only the control of the power for the actuation systems on the wing trailing edge, it was representative of the power control approach which was used for all flight control actuation functions (i.e. all those listed in Table 8). Figure 35 was limited to this coverage to avoid excess complexity in the presentation and thus to avoid confusion which such complexity was likely to generate.

Figure 37 shows the general arrangement of the signal and power hookups between power systems, flight data computers, and the various microprocessors.

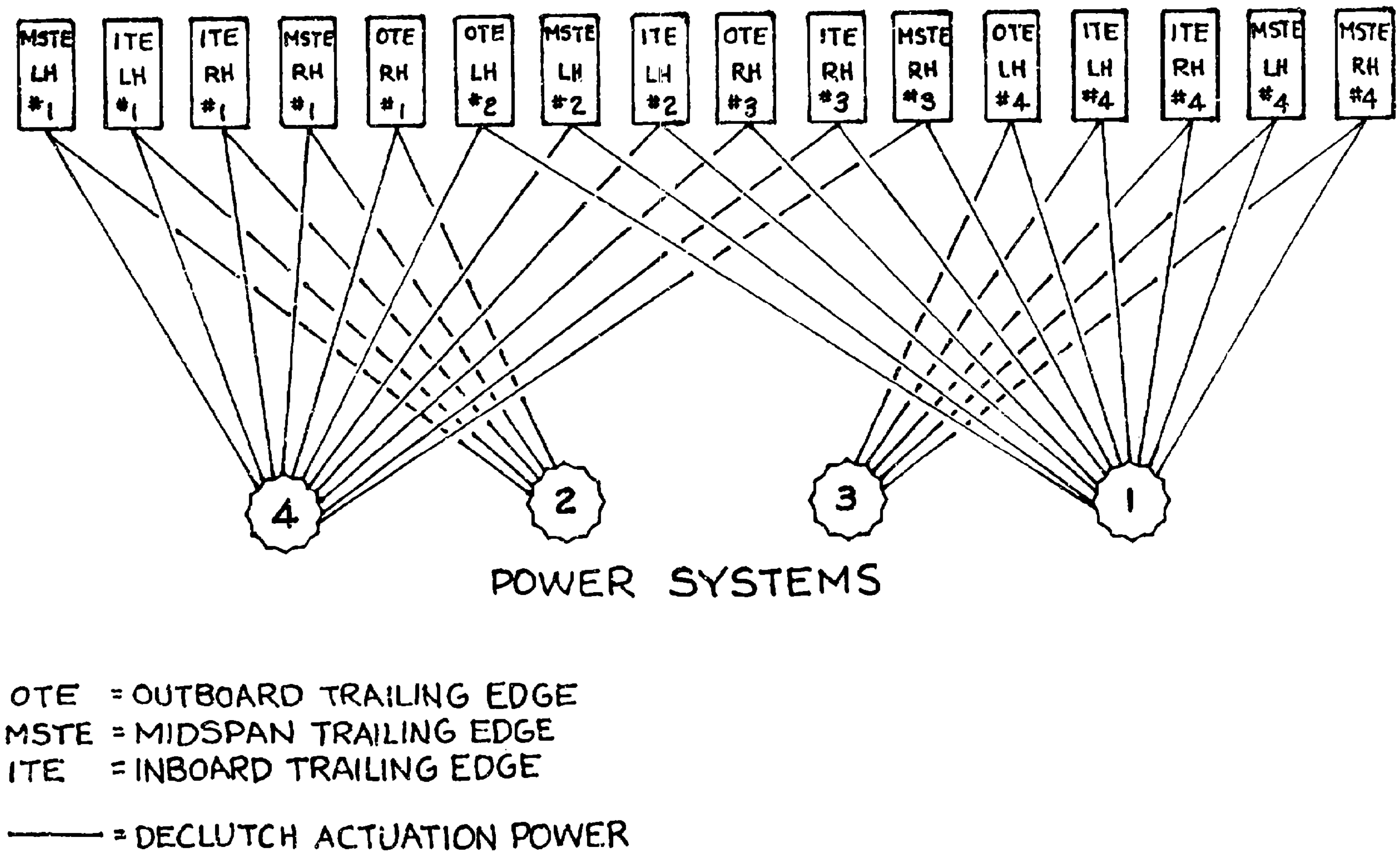


Figure 36. Wing Trailing Edge System Actuator Declutching Connections

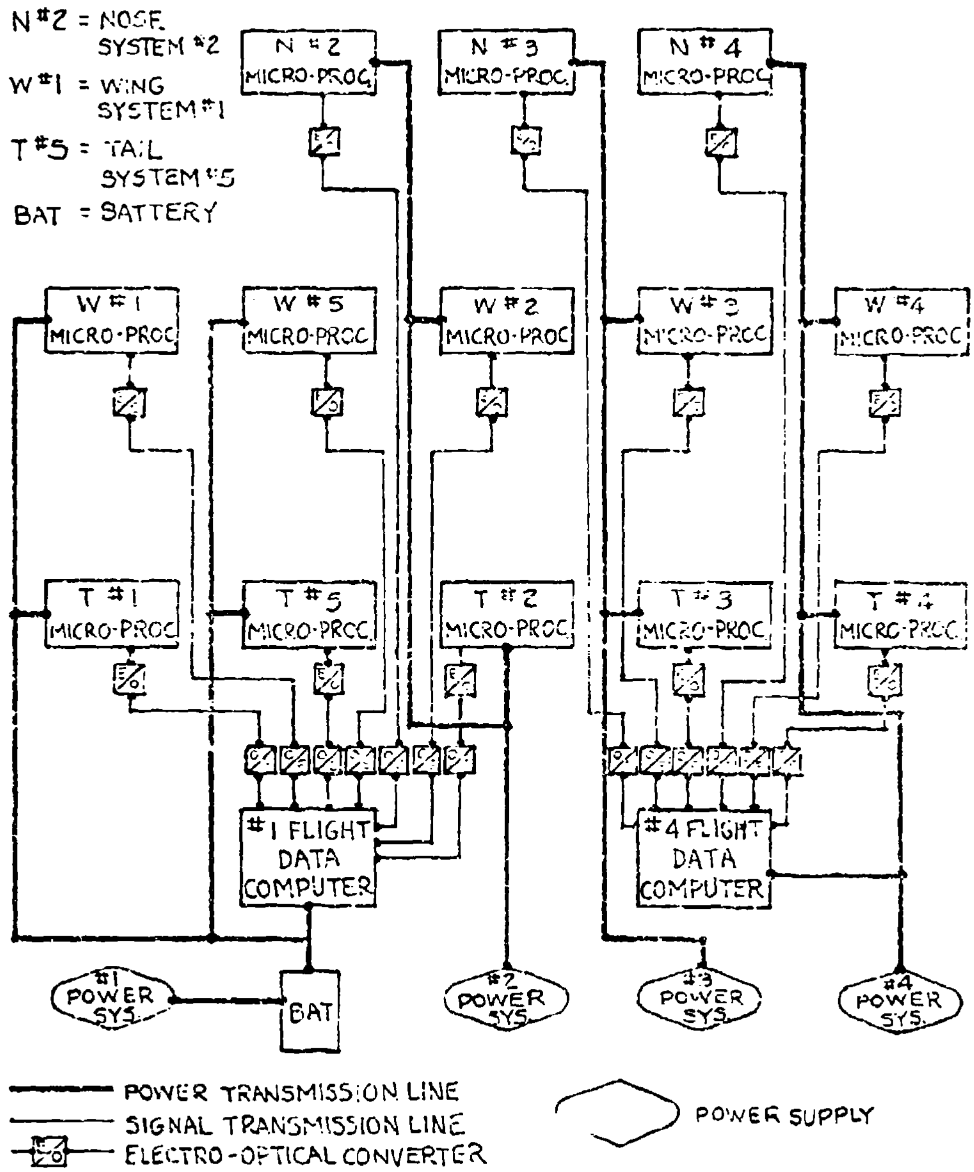


Figure 37. Signal and Power Hookups

The figure shows that there are two redundant flight data computers. The first one, powered by system No. 1, was located in the aft avionics compartment (see Figure 22) and the second one, powered by system No. 4 was located in the intermediate avionics compartment. This gave wide system separation between the two computers and greatly reduced the possibility that battle damage would incapacitate the two units simultaneously. It can be seen in Figure 37 that the microprocessors were divided into 3 geographical groups,nose,wing and tail, which indicates the general physical location of the microprocessors in the aircraft as well as the actuation functions which they serve. W #1 microprocessor and W #2 microprocessor in Figure 37 were the same as system #1 microprocessor and system #2 microprocessor respectively in Figure 35. W #5 microprocessor in Figure 37 was the "model" microprocessor indicated but not shown in Figure 35. The signal transmission interconnects shown in Figure 37 between the #1 and #4 flight data computers and the wing (W) grouping of microprocessors were the same interconnects as those indicated as "flight control input channels" on Figure 35. It can also be seen in Figure 37 that the "wing" and "tail" groups contained full five channel capability with five microprocessors each whereas the nose group contained only three microprocessors. The reason for this was the fact that the wing and tail microprocessor groups service all the flight critical (2 fail operate-fail safe) functions on the aircraft. These were also the functions which could not tolerate a hard over signal after the third failure nor could they tolerate being locked in any position other than trail after a third failure. To accomplish this it was necessary to have five signal channel capability so that a defective signal channel could be voted out of action as a result of its being the third failure.

In the case of the nose grouping, however, circumstances were considerably different. Here only 3 functions required the services of microprocessor controlled inverters. These were the gun, the nose gear steering and the canard. Of the three only the canard had greater than a fail safe requirement. The canard was single fail operate-fail safe, however, because the nature of the canard was such that it could and would, aerodynamically blow back and lock when disconnected. The disagreement betweeen channels which would exist after a second failure could be used to trigger a disconnect and, therefore, no more than 3 channels were required to meet fail safe requirements.

4.1.6 Starter/Generator System Definition - A more detailed definition of the starter/generator system, was prepared which expanded on that given in paragraphs 4.1.1 and 4.1.2. This improved definition was derived from the additional data generated in paragraphs 4.1.4 and 4.1.5. The general characteristics of the starter/generator system thus defined were as follows:

Generator Rating:	60/70 KW AC/DC output
AC Output:	115/200 V 400 Hz 3 phase power per MIL-STD-704A
DC Output:	270 VDC per MIL-STD-704A
Speed (Generating):	Range = 2:1 Minimum = 13750 RPM Maximum = 27500 RPM
(Starting):	Minimum = 0 RPM Maximum = 10,000 RPM
Cycloconverter Rating:	20 KW (25 KVA at 0.8 PF)
Transformer-Rectifier-Filter Rating:	47.5 KW
Cooling:	Conduction (cold plate) oil cooling-cool oil supplied by AMAD (i.e. shared system)

Weight:

ITEM	ITEM CODE (Figure 4.2)	WEIGHT LBS	REF CODE
General Equipment			
Generator	(G)	59.2	1
Cycloconverter	(CCV)	30.7	2
Transformer-Rectifier-Filter	(TRF)	17.1	3
Generator Control Unit	(GCU)	8.0	4
Generator System Total		115.0	
Starting System Equipment (For Starter/Generator Only)			
Drain and Fill Torque Converter		6.1	5
90 KW Reverse SCRs and Control	(RSCR)	21.5	5
15 KW Inverter Unit	(SI)	19.8	5
90 KVA AC Start Contactor	(SCA)	6.5	5
15 KVA AC Start Contactor	(SCB)	0.7	5
15 KW DC Start Contactor	(SCC)	8.0	5
90 KW DC Start Contrctor	(SCD)	9.0	5
Starting System Equipment Total		71.6	
Generating System Total		115.0	
Starter/Generator System Total		186.6	

The generator, cycloconverter, and transformer-rectifier-filter ratings listed above were derived from the electrical load analysis summation, listed in Table 4-3, using the following logic. From Table 4-3, assuming emergency conditions with two systems failed (2 channels operative) and considering the most highly loaded of the two remaining channels (i.e. channel 2), the maximum required outputs of the generating system were as follows: (Table 4-3 values rounded off)

AC Output (continuous load)	20 KW (25 KVA at 0.8 PF)
DC Output (continuous load)	35 KW
AC Output (5 second load)	20 KW (25 KVA at 0.8 PF)
DC Output (5 second load)	95 KW

From these values the following ratings of the cycloconverter and transformer-rectifier-filter were derived.

Cycloconverter (based on continuous loads)	20 KW (25 KVA at 0.8 PF)
Transformer-rectifier-filter (based on 50% of 5 sec load)	47.5 KW

From the foregoing ratings the generator ratings were derived as follows:

Continuous load rating
20 + 47.5 = 67.5 KW rounded off for growth = 70 KW

5 second load rating
95 + 20 = 115 KW rounded off to 120 KW X 50% = 60 KW

Hence the generator was assigned a 60/70 KW rating.

The weights for the various starter/generator system elements were derived from various data source references as indicated by the "REF CODES" used in the right hand column of the weight tabulation above and listed below.

1. Average of the data from two sources i.e., reference 11 page 54 and reference 14 page 39.

2. Data from reference 14 reduced by 10% to account for advances in the 1990+ time period

3. Derived from a curve plotted from data from reference 11 page 38

4. Data from reference 11 page 38 through 41

5. Data from reference 11 page 39

To arrive at the generating system defined above, three generating system approaches were considered all of which had the following characteristics in common.

1. Each was rated at 60/70 KW output

2. Two of the four generating systems, one on each engine, had engine start capabilities.

3. Each was capable of delivering 20 KW (25 KVA at 0.8 PF) of continuous AC power and 47.5 KW of continuous DC power. The three generating system approaches considered were:

   1. Integrated starter generator (ISD) employing a constant speed drive.

   2. VSCF starter/generator system employing a cycloconverter for AC output.

   3. VSCF starter/generator system employing a DC link for AC output.

Although the starting system based upon the ISD type starter/generator was the lightest by approximately 58 lbs, it was dropped from consideration because its full load and cruise load efficiency was poorer than that of the other two approaches by approximately 14% (Reference 28 page 52) and because of its relatively poor reliability. The poor cruise efficiency effectively cancelled a large portion of its weight advantage (i.e. 47 lbs) as shown in the following analysis:

The sum of the average powers delivered at the four busses is 80 KW during a typical mission. Based on this, the power extracted from the 2 jet engines combined is:

For ISD system $\dfrac{80}{0.71 \text{EFF}} = 112.68$ KW or 151.05 H.P.

For VSCF system $\dfrac{80}{0.85 \text{ EFF}} = 94.12$ KW or 126.16 H.P.

Therefore the ISD system extracts 24.89 HP more than the VSCF system.

Assuming a specific fuel consumption of 0.7 lb fuel/BHP-HR for the jet engines and using the 2.7 HR maximum unrefueled mission time of the ATS aircraft (page 6), the total

extra fuel required for the aircraft using the ISD is:

24.89 HP X 0.7 lb/HP-HR X 2.7 HR = 47.04 lb

The ISD's relatively poor reliability has been recognized for years and has been the driving force behind the development of the VSCF approach. In view of it's small potential weight saving and it's poor record in maintainability and reliability, the ISD was not considered a viable candidate for the 1990+ time period and was dropped from further consideration.

There was very little to choose between the cycloconverter and DC link approaches to VSCF generator design. At the size (60/70 KW) and speed range 2:1) characteristic of this application it was projected that the cycloconverter would be 5 to 10% lighter than the DC link (reference 28) for the same power output. However, the DC link full load efficiency would be 5-9% greater than that of the cycloconverter and its temperature tolerance would be greater (120°C vs 80°C continuous input cooling oil temperature limit). Balanced against this the cycloconverter's part load effeciency was 1-2% better than the DC link (reference 18). Because cooling heat critical components would not be a problem in this aircraft (evaporative cooling) the heat tolerance advantage of the DC link was considered offset. Therefore, because the generator operated at part load most of the time and because weight was critical, the cycloconverter approach was selected.

4.1.7 APU Driven Generator Sizing - As shown in Figure 17 two generators, rated at 45KW each, are driven by the APU. This size selection was justified as part of the discussions in paragraph 4.1.2.3. Based on this power rating each generator including its generator control unit (GCU) plus its 45 KW transformer rectifier (TRF) and 7 KVA cycloconverter (CCV), weighed 83.9 lbs.

4.1.8 APU Sizing - The APU for aircraft II was rated at 485 HP sea level static and weighed 245 lbs including all peripherals such as a starter, fuel control, blade containment provisions, reduction gearbox with generator mounting pads, lube oil, oil tank, etc.. The APU was a free turbine unit with an annular inlet, three-stage axial and single-stage centrifugal compressor, annular combustion chamber, single-stage compressor-turbine and counter rotating power turbine and is similar in functional arrangement to the Hamilton Standard ST61-73 APU.

The power rating determination was based on the following computation:

89.5 KW required at primary generator shaft (Paragraph 2.2.7)

÷ 0.85 Primary generator efficiency

105.29 KW required at primary generator terminals

÷ 0.97 Transmission efficiency (voltage drop)

108.55 KW required at APU generator output terminals for starting load (See comparative value at the end of paragraph 4.1.4)

+50.28 KW electrical system 5 sec essential loads from Table 16 sht 12

158.83 KW total required at APU generator output terminals during starting

÷ 0.85 APU generator efficiency

186.86 KW at APU generator shaft

÷ 0.94 APU adapter gearbox efficiency

198.79 KW required APU output at 20,000 ft

÷ 0.55 Sea level correction factor (reference 25 page 12)

361.43 KW sea level static rating

÷ 0.746 KW to HP conversion factor

484.49 HP (use 485 HP rating)

The APU had a 0.610 lb/HP-HR specific fuel consumption during typical starting and emergency return duty cycles and fitted in a rectangular compartment whose dimensions were 18 X 18 X 40 in.

4.1.9 AIRCRAFT I ELECTRICAL SYSTEM WEIGHT ANALYSIS-

The Aircraft I weight analysis considered all those elements of the electrical system which were unique to, or were otherwise impacted by, the change from the more conventional secondary power system arrangement used in Aircraft II (see paragraph 4.2) to the "ALL ELECTRIC" approach used in Aircraft I. In effect, therefore, Aircraft II became the "BASELINE" aircraft against which all other variants were measured. The major elements considered in arriving at Aircraft I's relative weight were as follows:

1. All the electrically powered actuation functions which were hydraulically powered in Aircraft II.

2. All the power distribution elements which service the actuation functions of item (1) above.

3. All the elements in the electrical power generation system.

4. All the components making up the auxiliary power and starting system.

5. The impacts on the fuel system resulting from changed heat rejection and fuel displaced by inverters.

Structural impacts were not considered since all actuators, for both aircraft, used the same tie off points and reacted the same loads. Minor impacts due to differences in actuator envelopes, actuator weights, bulkhead penetration points for electrical cables vs hydraulic lines, component weights and component envelopes were ignored as being so small as to be within the "Noise Level." Component weight and envelopes were, of course, considered in terms of growth factors and fuel displaced respectively.

Environmental control system impacts were not considered since, as discussed in paragraph 2.1.6, the systems, and heat loads they must service, were essentially identical between Aircraft I and Aircraft II.

4.1.9.1 Electrically Powered Actuation Functions - Table 25 lists the actuation functions outlined in item 1 above and shows the weight chargeable to each of these functions. The weight of the various components (i.e. motors, ballscrews, inverters, relays, etc.) making up each actuation subsystem, as shown in Table 25 was derived from data included as figures 38 and 39 and as derived or extrapolated from Table 21. Table 25 shows that the total weight of the actuation subsystems is 1667.3 lb.
The calculations used to determine the weight entries for the various ball screw actuator entries in Table 25 are presented as follows:

EXTERNAL FLAP

Stall Load	=	8600 lb (1)
Stroke	=	10 inch (1)
Motor Power	=	8.80 kw (2)

TABLE 25. ELECTRICAL ACTUATION FUNCTION WEIGHT

FUNCTIONAL ITEM	RATED POWER (KW)			MOTOR INPUT (AMPS)	ITEM WEIGHT (LB)	INVERTER WEIGHT (LB)	SSPC* WEIGHT (LB)	TOTAL WEIGHT (LB)	NUMBER OF ITEMS PER A/C	TOTAL WEIGHT PER A/C
	ACTUATOR OUT	INPUT TO MOTOR	MOTOR POWER							
MOTORS - DC INVERTER CONTROLLED										
INBOARD FLAP	11.16	18.60	33.57	68.8	12.0	38.0	—	50.0	6	300.0
MIDSPAN FLAP	2.27	3.72	6.71	13.8	2.3	12.5	—	14.8	6	88.8
AILERON	0.85	1.42	2.52	5.3	1.0	10.0	—	11.0	4	44.0
UPPER RUDDER	0.41	0.68	0.75	2.5	0.3	10.0	—	10.3	6	61.8
LOWER RUDDER	0.39	0.65	1.50	2.4	2.4	10.0	—	12.4	6	74.4
L.E. FLAP	0.75	2.10	3.70	7.8	2.6	10.0	—	12.6	12	151.2
PLUG THROAT	25.52	33.73	60.88	124.9	21.8	67.5	—	89.3	2	178.6
N.G. STEERING	0.43	0.56	1.01	2.1	1.4	10.0	—	11.7	1	11.7
MOTORS - DC RELAY CONTROLLED										
MAIN GEAR BRAKES	1.91	3.74	6.77	13.85	2.3	—	0.8	3.1	2	6.2
THRUST REVERSER	19.64	26.87	48.63	99.52	17.4	—	1.8	19.2	2	38.4
BALL SCREW ACTUATORS - DC MOTOR - INVERTER CONTROLLED										
CANARD	0.17	0.27	1.00	1.00	15.0	10.0	—	25.0	2	50.0
EXTERNAL FLAP	4.13	4.86	8.80	18.00	24.8	12.5	—	37.3	4	149.2
THRUST VECTOR VANE	0.72	1.13	2.05	4.19	167.7	10.0	—	177.7	2	355.4
BALL SCREW ACTUATORS - AC MOTOR - RELAY CONTROLLED										
R. RAM AIR SCOOP	0.93	1.09	1.97	4.04	8.9	—	0.5	9.4	1	9.4
L. RAM AIR SCOOP	0.01	.02	0.04	0.07	1.5	—	0.1	1.6	1	1.6
BALL SCREW ACTUATORS - DC MOTOR - RELAY CONTROLLED										
NOSE GEAR	1.40	1.65	2.99	6.11	34.4	—	0.6	35.0	1	35.0
MAIN GEAR	2.71	3.19	5.77	11.81	54.9	—	0.8	55.7	2	111.4
									TOTAL	1667.1

* RELAY

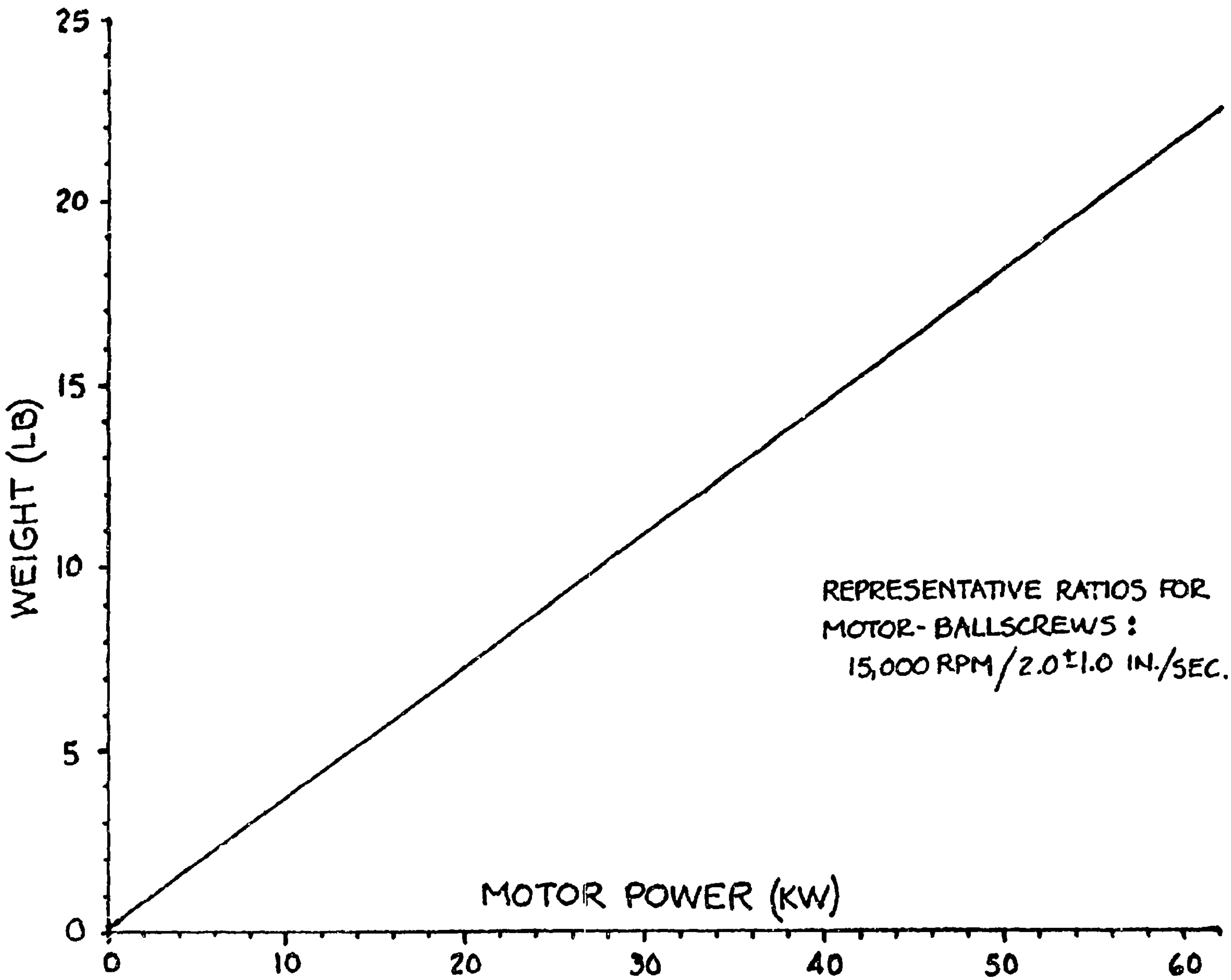


Figure 38. Motor Weight

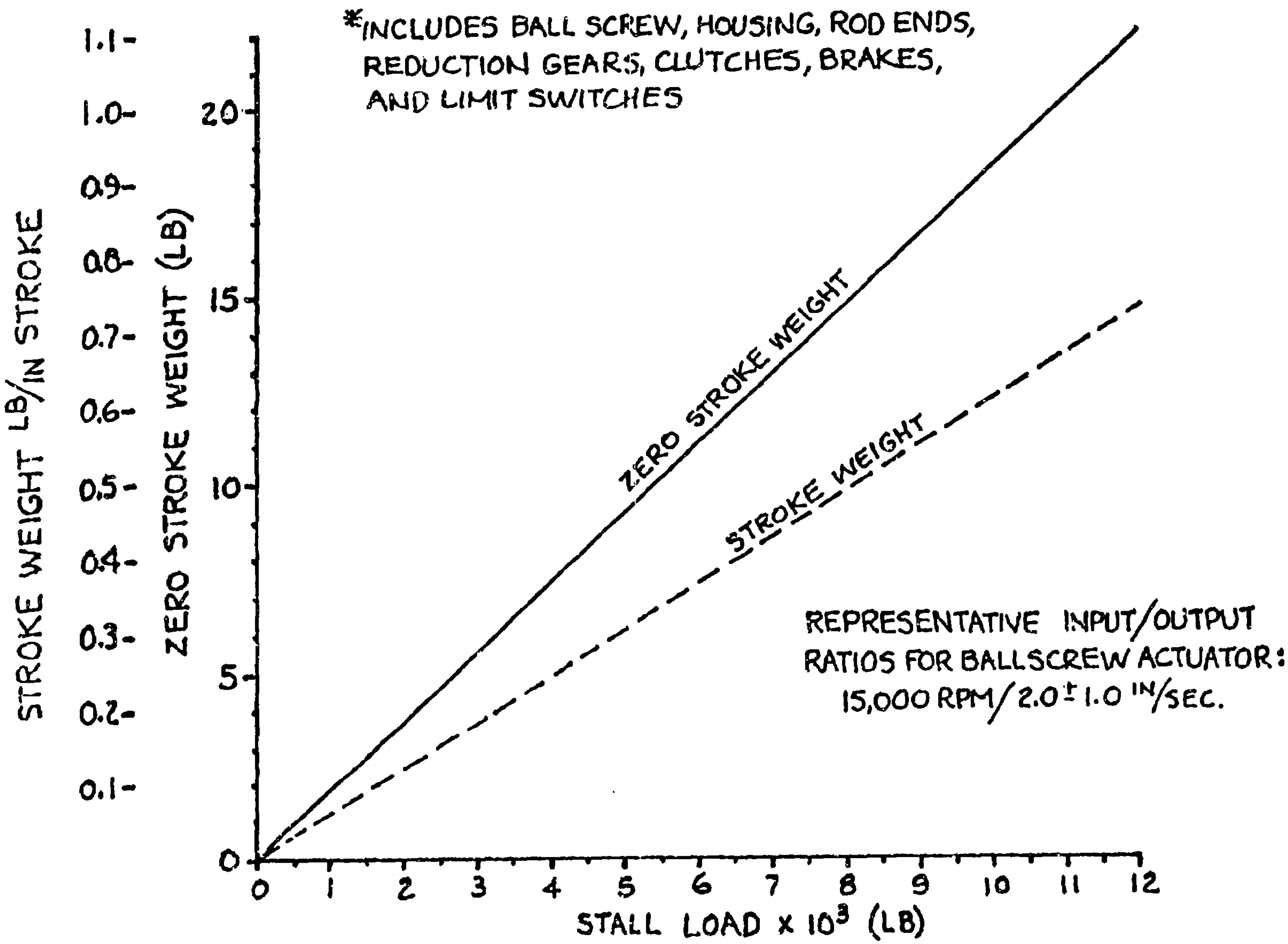


Figure 39. Ballscrew Actuator *Assembly Weight

4.1.9.1 (Cont.)

Weight

Zero Stroke Weight	= 16.2 lb	3
Stroke Weight (10 x .53)	= 5.3 lb	3
Motor	= 3.3 lb	4
Total	= 24.8 lb	

CANARD

Stall Load	= 6400 lb	5
Stroke	= 5.75 in	5
Motor Power	= 1.00 kw	2

Weight

Zero Stroke Weight	= 11.8 lb	3
Stroke Weight (5.75 x .39)	= 2.2 lb	3
Motor = (2 x 0.5)	= 1.0 lb	4
Total	= 15.0 lb	

THRUST VECTOR VANE

Stall Load	= 75,314 lb	6
Stroke	= 5.20 in	6
Motor Power	= 2.05 kw	2

Weight

Zero Stroke Weight	= 138.6 lb	7
Stroke Weight (5.2 x 4.59)	= 23.9 lb	7
Motor = (2 x 2.6)	= 5.2 lb	4
Total	= 167.7 lb	

RIGHT RAM AIR SCOOP

Stall Load	= 4100 lb	1
Stroke	= 2.52 in	1
Motor Power	= 1.97 kw	2

Weight

Zero Stroke Weight	= 7.5 lb	3
Stroke Weight (2.52 x .25)	= 0.6 lb	3
Motor Weight (1.0 x 0.8)	= 0.8 lb	4
Total	= 8.9 lb	

LEFT RAM SCOOP

Stall Load	= 1200 lb	1
Stroke	= 1.00 in	1
Motor Power	= .04 kw	2

Weight

Zero Stroke Weight	= 1.2 lb	3
Stroke Weight (1.0 x .08)	= 0.1 lb	3
Motor (1.0 x 0.2)	= 0.2 lb	4
Total	= 1.5 lb	

4.1.9.1 (Cont.)

NOSE GEAR

Stall Load	=	14,900 lb	1
Stroke	=	5.0 in	1
Motor Power	=	2.99 kw	2
Weight			

Zero Stroke Weight	=	27.4 lb	7
Stroke Weight (5.0 x .91)	=	4.6 lb	7
Motor Weight (2 x 1.2)	=	2.4 lb	4
Total	=	34.4 lb	

MAIN GEAR

Stall Load	=	24,000 lb	1
Stroke	=	6.00 in	1
Motor Power	=	5.77 kw	2
Weight			

Zero Stroke Weight	=	44.2 lb	7
Stroke Weight (6 x 1.44)	=	8.6 lb	7
Motor Weight (1 x 2.1)	=	2.1 lb	4
Total	=	54.9 lb	

1 See Table 9
2 See Table 25
3 See Figure 39
4 See Figure 38
5 See Figure 48 and Paragraph 4.2.1.6
6 See Table 12
7 Extrapolated from Figure 39

4.1.9.2 Electrical Power Distribution Elements - Figure 40 is a plain view of the aircraft showing the general location of the major components constituting system No. 1 power generation distribution and utilization elements. It was used as a basis for determining the wire lengths and sizes of the wiring used in the bus feeder and power distribution portions of system No. 1. It was assumed that system No. 1's lengths, sizes, and routings were sufficiently like the other three systems so that it could be considered a representative average of the other three. Thus, the total system's wiring weight was determined by multiplying the weight determined for system No. 1 by four. Table 26 is a detailed listing of the feeder and power distribution wiring for Aircraft I. It shows that the total weight of the power wiring including supports, harness, shielding and connectors was 120.3 lb. It should be remembered that this was the weight for 270 volt power distribution wiring and did not include 400 Hz AC power wiring, avionics equipment wiring, 28 VDC wiring, or the wiring for the fly-by-wire/fly-by-light system. In these latter four instances, as has already been discussed, Aircraft I and Aircraft II were considered essentially identical and thus these elements did not enter into the trade study.

4.1.9.3 Auxiliary Power and Starting System - The APU has already been defined in paragraph 4.1.8 and the starting system in paragraph 4.1.6. However, a major element of the auxiliary power system, not yet considered, was the battery which was provided primarily to supply power for a descent to 20,000 from any higher altitude in the event of an emergency. As pointed out in paragraph 4.1.2.3, this battery was to be of sufficient size to provide at least 4 minutes of power in an emergency descent mode. The continuous load,

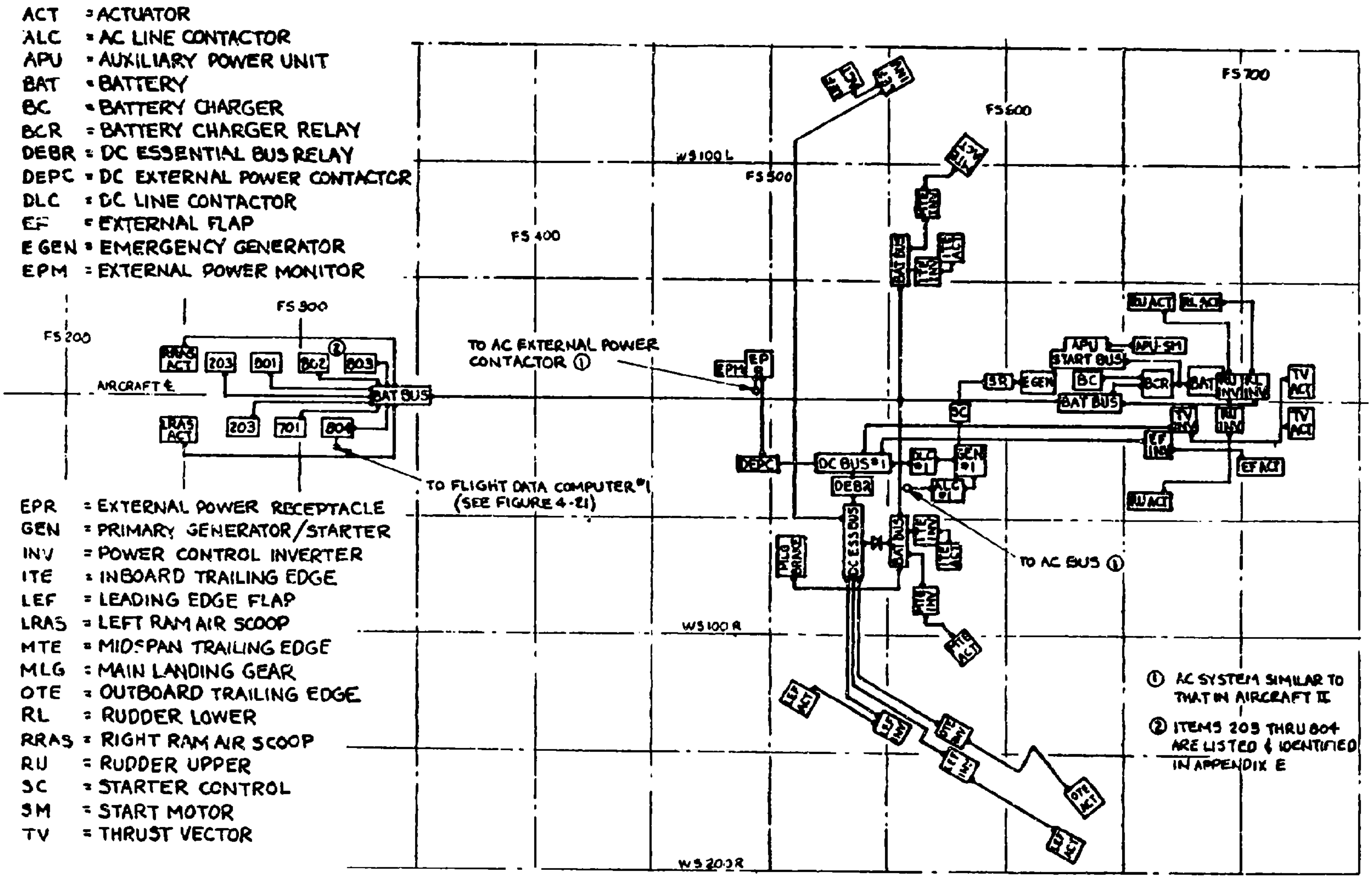


Figure 40. Electrical Component Locations

TABLE 26. WEIGHT - ELECTRICAL POWER WIRING (PAGE 1 OF 2 PARTS)

ROUT-ING NO.	SEG-MENT NO.	LEN-GTH FT.	RAT-ING AMP.	AWG NO. WIRES -GAGE	WT. LBS.
1	1-1	2	130	6 - 12	.28
2	2-2	2	130	6 - 12	.28
3	3-3	6	16	1 - 14	.09
	3-4	6	16	1 - 14	.09
	3-5	5	16	1 - 14	.07
4	4-6	6	6	1 - 20	.03
	4-7	6	6	1 - 20	.03
	4-8	6	6	1 - 20	.03
5	5-9	5	130	6 - 12	.70
	5-10	4	130	6 - 12	.56
6	6-11	2	130	6 - 12	.28
7	7-12	2	130	6 - 12	.28
8	8-13	2	88	6 - 14	.18
9	9-14	6	3	1 - 24	.02
	9-15	5	3	1 - 24	.01
10	10-16	6	5	1 - 22	.02
	10-17	6	5	1 - 22	.02
11	11-18	6	5	1 - 22	.02
	11-19	4	5	1 - 22	.02
12	12-20	6	5	1 - 22	.02
	12-21	6	5	1 - 22	.02
	12-22	6	5	1 - 22	.02
	12-23	6	5	1 - 22	.02
13	13-24	3	5	1 - 22	.01
14	14-25	6	3	1 - 24	.02
	14-26	4	3	1 - 24	.01
15	15-27	6	5	1 - 22	.02
	15-28	3	5	1 - 22	.01
16	16-29	5	5	1 - 22	.02
17	17-30	6	50	5 - 18	.24
18	18-31	3	30	2 - 14	.09
19	19-32	6	30	2 - 14	.18
20	20-33	3	6	1 - 20	.02
21	21-34	6	6	1 - 20	.03
22	22-35	6	8	1 - 18	.05
TOTAL WEIGHT THIS COL.					3.79

ROUT-ING NO.	SEG-MENT NO.	LEN-GTH FT.	RAT-ING AMP.	AWG NO. WIRES - GAGE	WT. LBS.
23	23-36	6	88	6 - 14	.54
	23-37	5	88	6 - 14	.45
24	24-38	6	88	6 - 14	.54
25	25-39	6	5	1 - 22	.02
	25-40	4	5	1 - 22	.02
26	26-41	5	30	2 - 14	.15
27	27-42	4	88	6 - 14	.36
28	28-43	6	10	1 - 16	.06
29	29-44	4	10	1 - 16	.04
30	30-45	4	3	1 - 24	.01
	30-46	4	3	1 - 24	.01
31	31-47	5	3	1 - 24	.01
	31-48	4	3	1 - 24	.01
32	32-49	6	6	1 - 20	.03
33	33-50	5	16	1 - 14	.07
34	34-51	5	3	1 - 24	.01
35	35-52	6	36	3 - 16	.18
36	36-53	3	30	2 - 14	.09
37	37-54	6	30	2 - 14	.18
38	38-55	3	6	1 - 20	.02
39	39-56	6	6	1 - 20	.03
40	40-57	6	130	6 - 12	.81
41	41-58	6	158	7 - 12	.95
42	42-59	6	9	1 - 18	.05
	42-60	6	9	1 - 18	.05
	42-61	6	9	1 - 18	.05
	42-62	7	9	1 - 18	.06
43	43-63	3	3	1 - 24	.01
	43-64	6	3	1 - 24	.02
	43-65	6	3	1 - 24	.02
44	44-66	8	3	1 - 24	.02
45	45-67	6	3	1 - 24	.02
	45-68	6	3	1 - 24	.02
46	46-69	6	5	1 - 22	.02
	46-70	8	5	1 - 22	.03
TOTAL WEIGHT THIS COLUMN					4.96
TOTAL WEIGHT THIS PAGE					8.75

TABLE 26. WEIGHT - ELECTRICAL POWER WIRING (PAGE 2 OF 2 PAGES)

ROUTING NO.	SEGMENT NO.	LENGTH FT.	RATING AMP	AWG NO. WIRES -GAGE	WT. LBS.
47	47-71	6	3	1-24	.02
	47-72	7	3	1-24	.02
48	48-73	6	3	1-24	.02
	48-74	5	3	1-24	.02
49	49-75	5	3	1-24	.02
50	50-76	8	3	1-24	.02
	50-77	10	3	1-24	.03
	50-78	10	3	1-24	.03
	50-79	10	3	1-24	.03
	50-80	12	3	1-24	.03
	50-81	10	3	1-24	.03
	50-82	8	3	1-24	.02
51	51-83	3	3	1-24	.01
	51-84	6	3	1-24	.02
	51-85	6	3	1-24	.02
1S	1S-86	2	130	6-12	.28
2S	2S-87	5	130	6-12	.70
3S	3S-88	6	130	6-12	.80
	3S-89	6	130	6-12	.80
	3S-90	6	130	6-12	.80
4S	4S-91	6	130	6-12	.80
	4S-92	7	130	6-12	1.00
1A	1A-93	3	160	7-12	.50
2A	2A-94	6	160	7-12	.90
1A	1A-95	3	160	7-12	.50
2A	2A-96	6	160	7-12	.90
TOTAL WEIGHT THIS COLUMN					8.32
TOTAL SYSTEM #1 WIRE WT.					17.07

TOTAL WIRE WEIGHT IN AIRCRAFT POWER SYSTEM (4 SYSTEMS) = 4 X 17.07 = 68.28

WEIGHT SUMMARY

TOTAL WIRE WEIGHT = 68.28

SUPPORTS, HARNESS & SHIELDING = 10.24

CONNECTORS-95/SYS AND 380/AC - AVERAGE WT. PER CONNECTOR = 1.1 LB.

380 X 1.1 = 41.80

TOTAL WIRING SYSTEM WEIGHT = 120.32

existing during an emergency descent, was determined as follows:

20.442 KW (See Table 16, PG 8, BAT BUS)

.810 KW (See Table 16, PG 5, ITEM 203)

2.268 KW (See Table 16, PG 6, ITEMS 701 through 804, BAT BUS

At the time this report was written, current state of the art NI-CAD batteries exhibited specific weights of 5.8 Watt-Hr/lb. This specific weight included such items, necessary for a practical installation, as the battery case, shrouding, thermistors and thermal switches. Because of the rapid advances which it was felt would have occured in battery technology by the 1990 time period, it was assumed that an equivalent battery specific weight at that time would be 11.4 watt-Hr/lb. From this, the battery weight was determined as follows:

$$\frac{23.520 \text{ Watts}}{11.5 \frac{\text{Watt-Hr}}{\text{LB}}} \quad X \quad \frac{4 \text{ Min}}{60 \frac{\text{Min}}{\text{HR}}} \quad X \quad 1.5 \text{ Operating Margin} = 204.52 \text{ Lb.}$$

Use 205 Lb

4.1.9.4 System Weight Summary - Table 27 summarizes all the weights subject to trade in the Aircraft I electrical system. As can be seen in Table 27 the total weight was 2817.0 Lbs.

TABLE 27. AIRCRAFT I ELECTRICAL SYSTEM WEIGHT SUMMARY

EQUIPMENT ITEM ①	QUANTITY PER A/C	UNIT WEIGHT (LB)	TOTAL WEIGHT (LB)
G - PRIMARY GENERATING SYSTEM ②	4	115.0	460.0
GE - EMERGENCY GENERATOR ③	2	83.9	167.8
APU - AUXILIARY POWER UNIT ③	1	245.0	245.0
STARTING SYSTEM EQUIPMENT ②	2	71.6	143.2
AEPC - AC EXTERNAL POWER CONTACTOR	4	1.9	7.6
ALC - AC LINE CONTACTOR	4	1.9	7.6
DEPC - DC EXTERNAL POWER CONTACTOR	4	2.5	10.0
DLC - DC LINE CONTACTOR	4	2.9	11.6
AEBR - AC ESSENTIAL BUS RELAY	2	2.0	4.0
BCR - BATTERY CHARGER RELAY	1	0.6	0.6
BR - BATTERY RELAY	1	2.0	2.0
BS - BATTERY SWITCH	1	0.2	0.2
B - BATTERY	1	205.0	205.0
DEBR - DC ESSENTIAL BUS RELAY	2	3.0	6.0
EPM - EXT. POWER MONITOR & RECPT. (AC)	1	3.0	3.0
EPM - EXT. POWER MONITOR & RECPT. (DC)	1	4.0	4.0
SR - STARTER RELAY	2	3.8	7.6
BC - BATTERY CHARGER	1	5.0	5.0
ACT - ELECTRICAL ACTUATION FUNCTION ④	—	—	1167.1
AMAD - AIRFRAME MOUNTED ACESSORY DRIVE	2	90.0	180.0
WIRE, SUPPORTS & HARNESS ⑤	—	—	78.5
CONNECTORS ⑤	—	—	41.8
COMPONENT SUPPORTS AND MISC.	—	—	59.4
		TOTAL WEIGHT	2817.0

① ACRONYMS HEADING ITEM TITLES CORRELATE WITH FIG.
② SEE PAGE ③ SEE PAGE ④ SEE TABLE
⑤ SEE TABLE

4.2 Aircraft II - Aircraft II represented the more conventional approach to secondary power generation, distribution, and utilization in that the aircraft used hydraulic power to power those components which have historically been powered electrically on advanced military aircraft of the immediate past. Thus aircraft II had a conventional power split between hydraulics and electrical but departed from the conventional by using an advanced (8000 PSI rated pressure) hydraulic system. The selection of 8000 PSI as the hydraulic system's rated pressure was based on extensive study programs and hardware development programs, as well several flight tests conducted at Rockwell's Columbus Division. This series of programs has been conducted over the last 15 years and has been documented in references 1 through 11. These programs indicated that 8000 PSI was very close to an optimum system pressure for advanced hydraulic systems given current and near future materials of construction. A listing of the major advantages and disadvantages of 8000 PSI rated system pressure, versus the current conventional 3000 PSI, as derived from these programs follows:

ADVANTAGES

1. Projected weight saving 30%
2. Projected installed volume saving of 40%
3. Survivability gains due to:
   - A. Less Volume
   - B. Less projected area
   - C. Heavier walled components
4. Lower component costs due to:
   - A. Less material used
   - B. Less machining costs due to heavier walls largely off setting the higher costs of slightly tighter tolerances

DISADVANTAGES

1. New ground test equipment required
2. Shear stable fluids required
3. Adverse effects of actuator stiffness

Table 28 represents a generic weight breakdown of the hydraulic system in the baseline study aircraft and showed the expected system weight at three rated operating pressures (3000, 4000 and 8000 PSI). This figure showed a projected weight saving of 584 lbs by using an 8000 PSI system in preference to a 3000 PSI system. Using a figure of 2.7 lb of gross take off weight saved per lb of direct weight saving the gross take off weight could be reduced by 1577 lb through the use of 8000 PSI and was considered a very significant figure.

Figure 41 is a plot of the data from Table 28 and showed that 8000 PSI was very close to an optimum weight i.e., that increases in operating pressure above 8000 PSI would not achieve significant further reductions in weight. This further verified that 8000 PSI was probably the proper pressure selection.

As will be seen elsewhere (paragraph 4.2.1.6) all flutter (stiffness) critical actuators on the study aircraft (i.e. the upper rudder, the aileron and the inboard flap) were powered through mechanical hinges using hydraulic motors. Since the gear reduction between motor and surface in all instances, was

TABLE 28. ATS HYDRAULIC SYSTEM WEIGHTS AT VARIOUS PRESSURES

	SYSTEM PRESSURE		
	3000 PSI	4000 PSI	8000 PSI
HYDRAULIC SYSTEM	[793]	[709]	[487]
POWER GENERATION	(358)	(319)	(251)
PUMPS	98	88	65
RESERVOIRS	117	103	76
SUPPORTS	7	6	4
MISCELLANEOUS	136	122	106
POWER DISTRIBUTION	(435)	(390)	(236)
PLUMBING & FITTINGS	263	235	142
FLUID	172	155	94
FLIGHT CONTROLS	[866]	[774]	[682]
ACTUATION DEVICES	(490)	(438)	(399)
CYLINDERS & ROTARY ACT.	454	406	377
MECHANISM & SUPPORTS	36	32	22
CONTROL MODULES	(275)	(247)	(228)
SERVO VALVES ETC.	275	247	228
POWER DISTRIBUTION	(101)	(89)	(55)
PLUMBING & FITTINGS	68	60	37
FLUID	33	29	18
LANDING GEAR & DOORS	[128]	[114]	[84]
ACTUATION DEVICES	(51)	(45)	(41)
CYLINDERS ETC.	43	38	36
MECHANISM & SUPPORTS	8	7	5
POWER DISTRIBUTION	(77)	(69)	(43)
PLUMBING & FITTINGS	66	59	36
FLUID	11	10	7
MISCELLANEOUS SYSTEMS	[159]	[143]	[109]
ACTUATION DEVICES	(92)	(82)	(74)
CYLINDERS, MOTORS ETC.	84	75	69
MECHANISM & SUPPORTS	8	7	5
POWER DISTRIBUTION	(67)	(61)	(35)
PLUMBING & FITTINGS	50	46	26
FLUID	17	15	9
TOTAL	1946	1740	1362

POWER GENERATION	358	319	251
POWER DISTRIBUTION	630	609	369
ACTUATION			
ACTUATORS	633	[illegible]	514
CONTROL MODULES	275	[illegible]	228

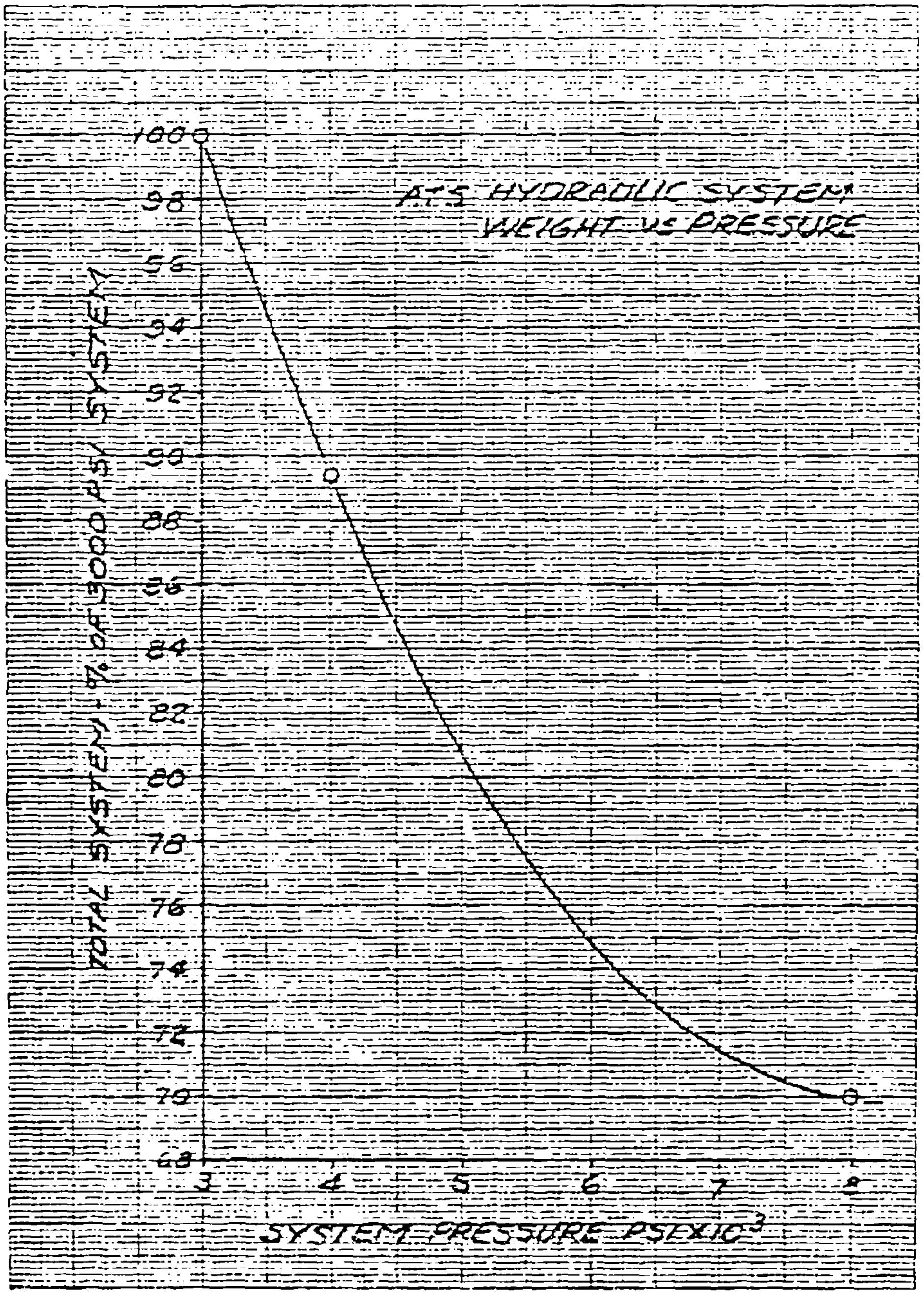


Figure 41. ATS Weight Trends at Various System Pressures

greater than 5000:1 (see Table 18 & 19), stiffness in the hydraulic circuit was no problem. With stiffness eliminated as a consideration the "advantages" listed above so far outweighed the "disadvantages" that 8000 PSI was selected as the design pressure.

A double voltage approach was also considered as a means to update and modernize the electrical power system on aircraft II in a manner similar to the update of the hydraulic system. However, for the reasons already given in paragraph 4.1.1, this approach did not seem to offer any advantage so a conventional 115/200V 400 HZ AC power system was retained for aircraft II.

4.2.1 Hydraulic System Description - The baseline hydraulic power generation, distribution, and utilization system is shown in block diagram form in Figure 42. Figure 43 is a more detailed schematic showing all major components making up the total hydraulic system, and Figure 44 shows the spatial arrangement of these components. The system had two equal authority hydraulic systems and a third emergency system. The third system powered only those functions necessary to recover from a maneuver and maintain level flight. The primary hydraulic system pumps were rated at 8000 psi. They were driven by airframe mounted accessory drives (AMADs), two pumps to a system, each powered by the same engine, and incorporating the master slave concept. A fifth pump, also rated at 28 GPM and 8000 PSI, was driven by the APU/EPU which acted as a third (emergency) power source. The AMAD was driven by the engines via a power take off shaft (PTO) or as an alternate by the auxiliary/emergency power unit (APU/EPU) shown in Figure 45. The APU/EPU was started by a hydraulic motor using stored energy from an accumulator.

4.2.1.1 APU/EPU Operation - The APU/EPU operation under various conditions was as follows:

A. APU/EPU Start - Aircraft on Ground - Engines Inoperative

APU/EPU startup is initiated by a cockpit switch. This switch, using battery power, actuates a solenoid valve which ports high pressure hydraulic fluid, stored in an accumulator, to the APU/EPU start hydraulic motor (SM in Figure 45). The motor accelerates the APU compressor - turbine to light-off speed. Using the combustion of a mixture of jet fuel and air in the conventional manner as an energy source, the compressor turbine accelerates to operating speed. At full speed the surplus energy from combustion (i.e. the energy over and above that necessary to power the compressor turbine) drives a free turbine which in turn, through suitable gearing, powers a 28 GPM (131 H.P.) "emergency" hydraulic pump, a 7 KW (9.4 H.P.) emergency generator and a 180 KW 242.0 H.P.) load compressor. It is capable of starting at all airport altitudes up to 5000 ft. (1524 M) as well as achieving inflight starts at all altitudes up to 20000 ft (6096M).

B. Engine Start - Aircraft on the Ground

1. Normal Start Using APU/EPU

With the APU/EPU running, the load compressor (see Figure 45) delivers pneumatic power to the air turbine start motor (ATS/M) associated with the engine selected for starting. The load

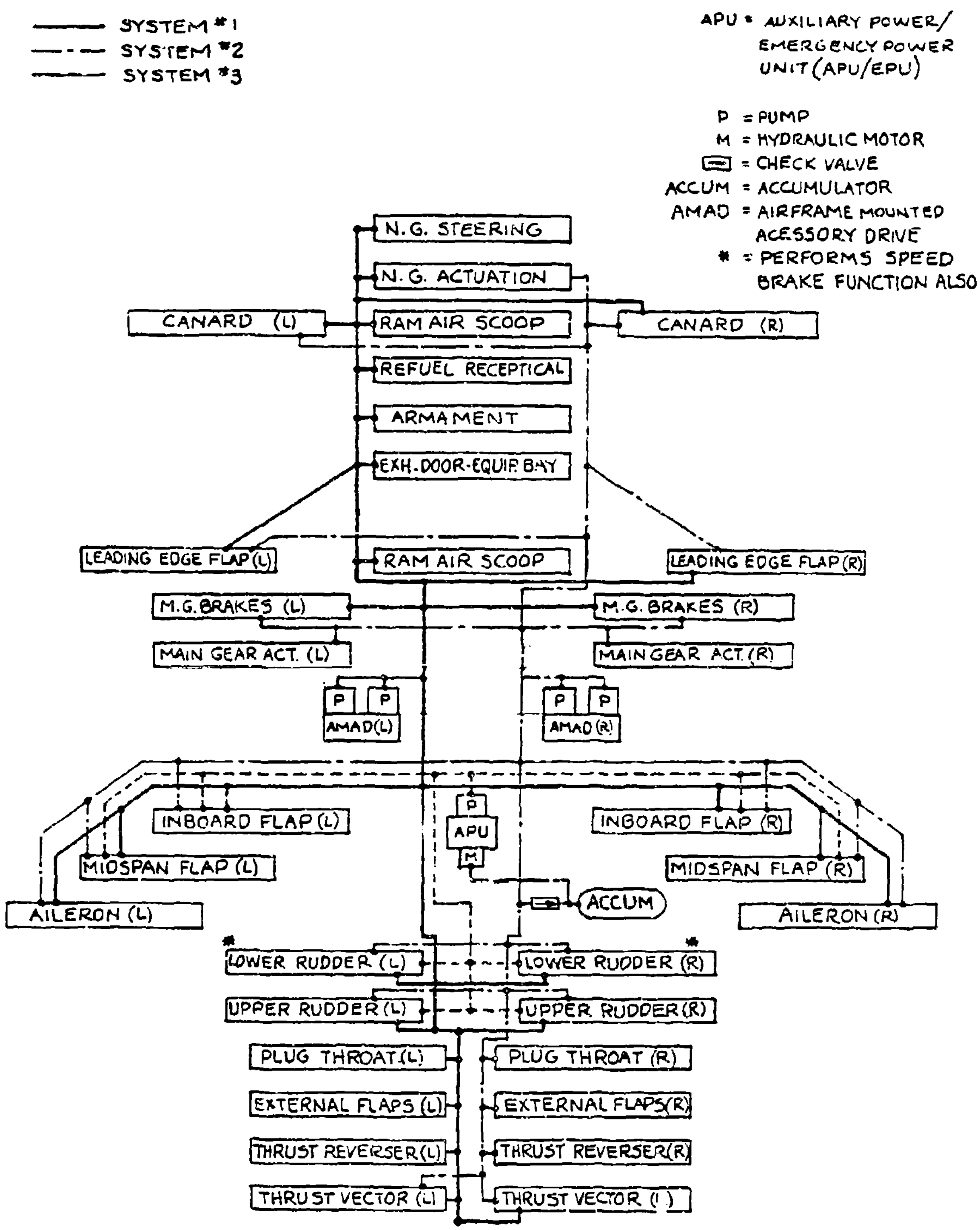
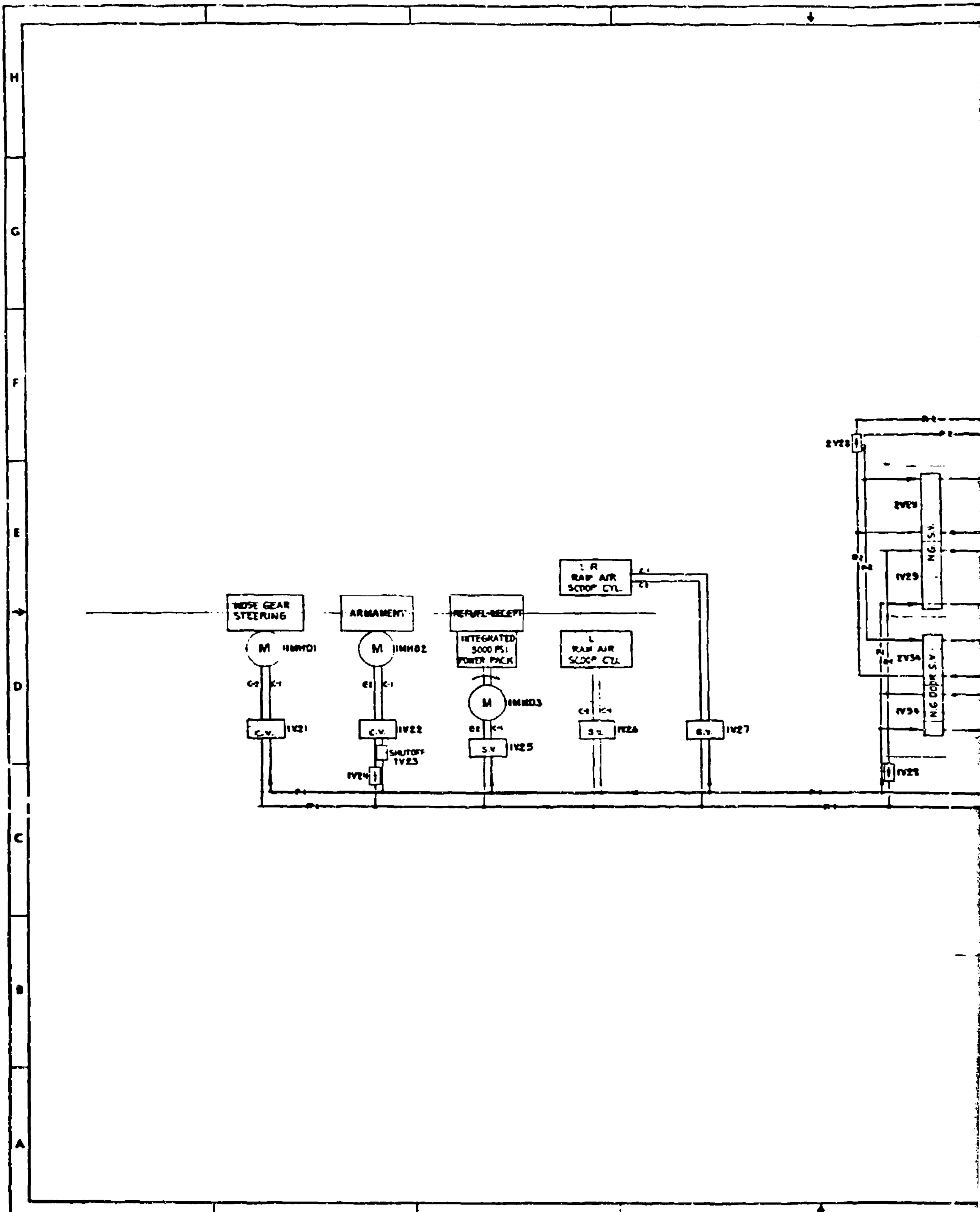
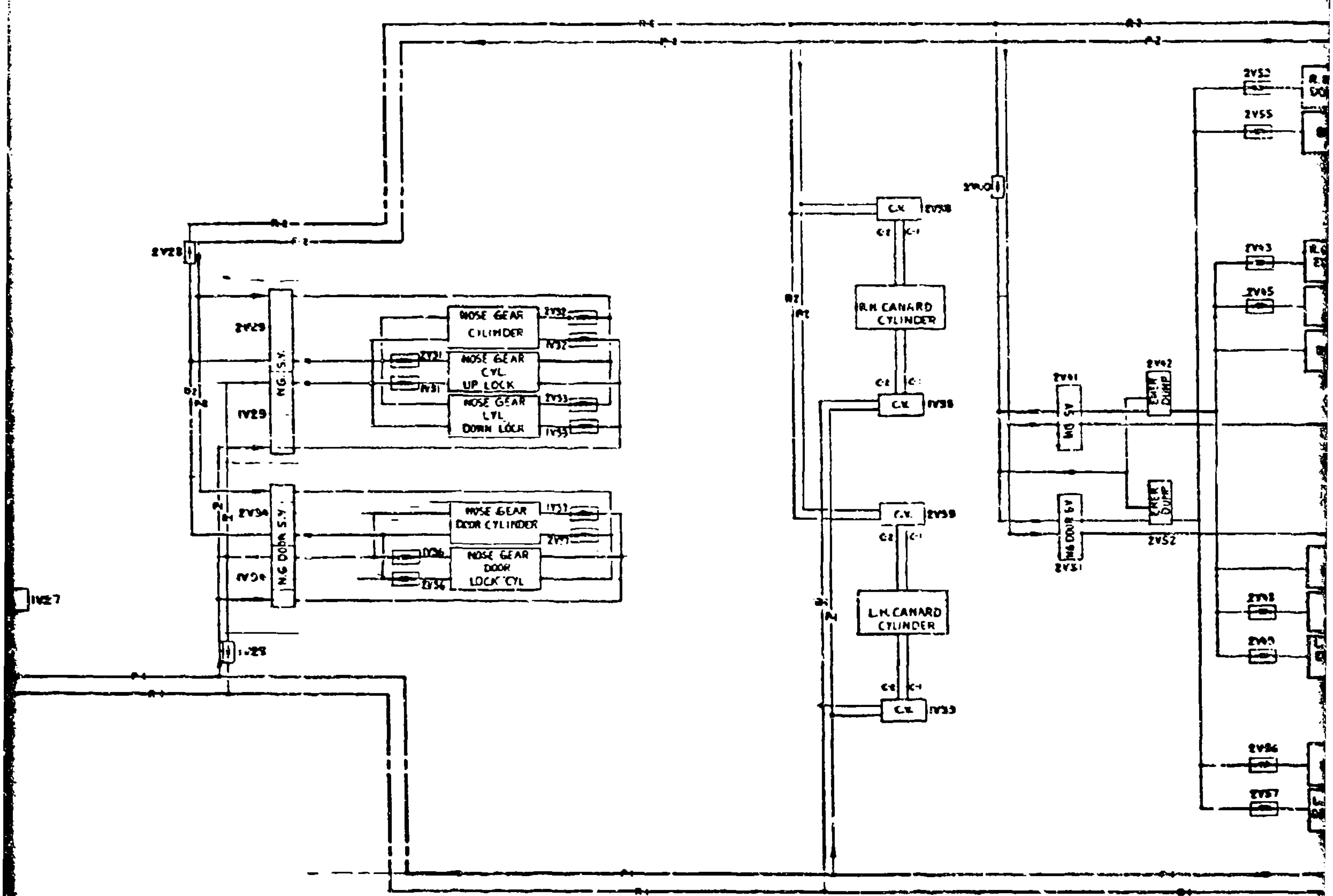
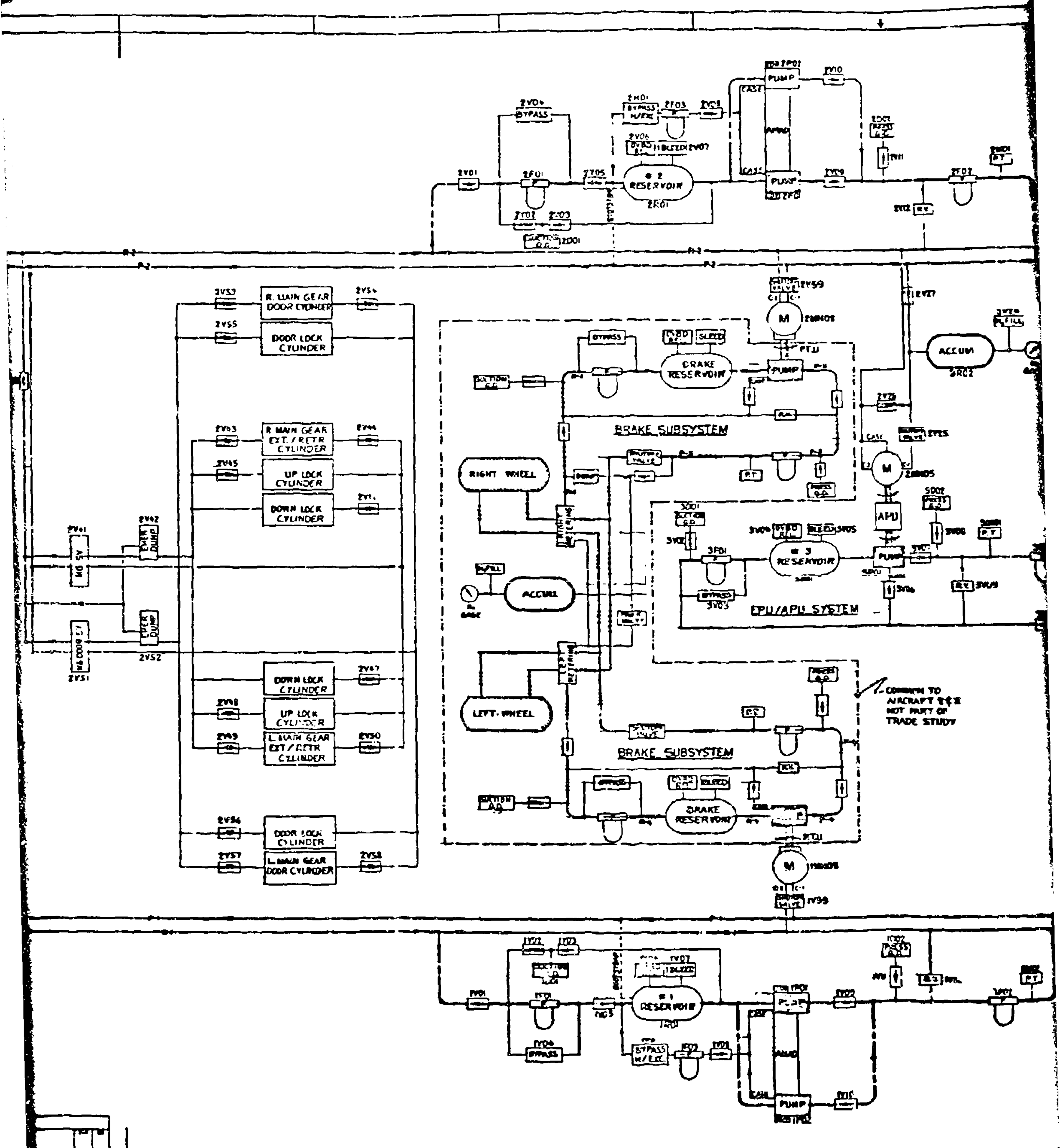


Figure 42. Hydraulic System Block Diagram


H
G
F
E
D
C
B
A
NOSE GEAR STEERING
ARMAMENT
INTEGRATED 3000 PSI POWER PACK
RAM AIR SCOOP CYL.
RAM AIR SCOOP CYL.
M
M
M
1MH02
1MH03
C.V.
C.V.
S.V.
B.V.
1V21
1V22
SHUTOFF 1V23
1V24
1V25
1V26
1V27
1V28
2V28
1V29
2V34
1V34
N.G. S.V.
N.G. DOOR S.V.

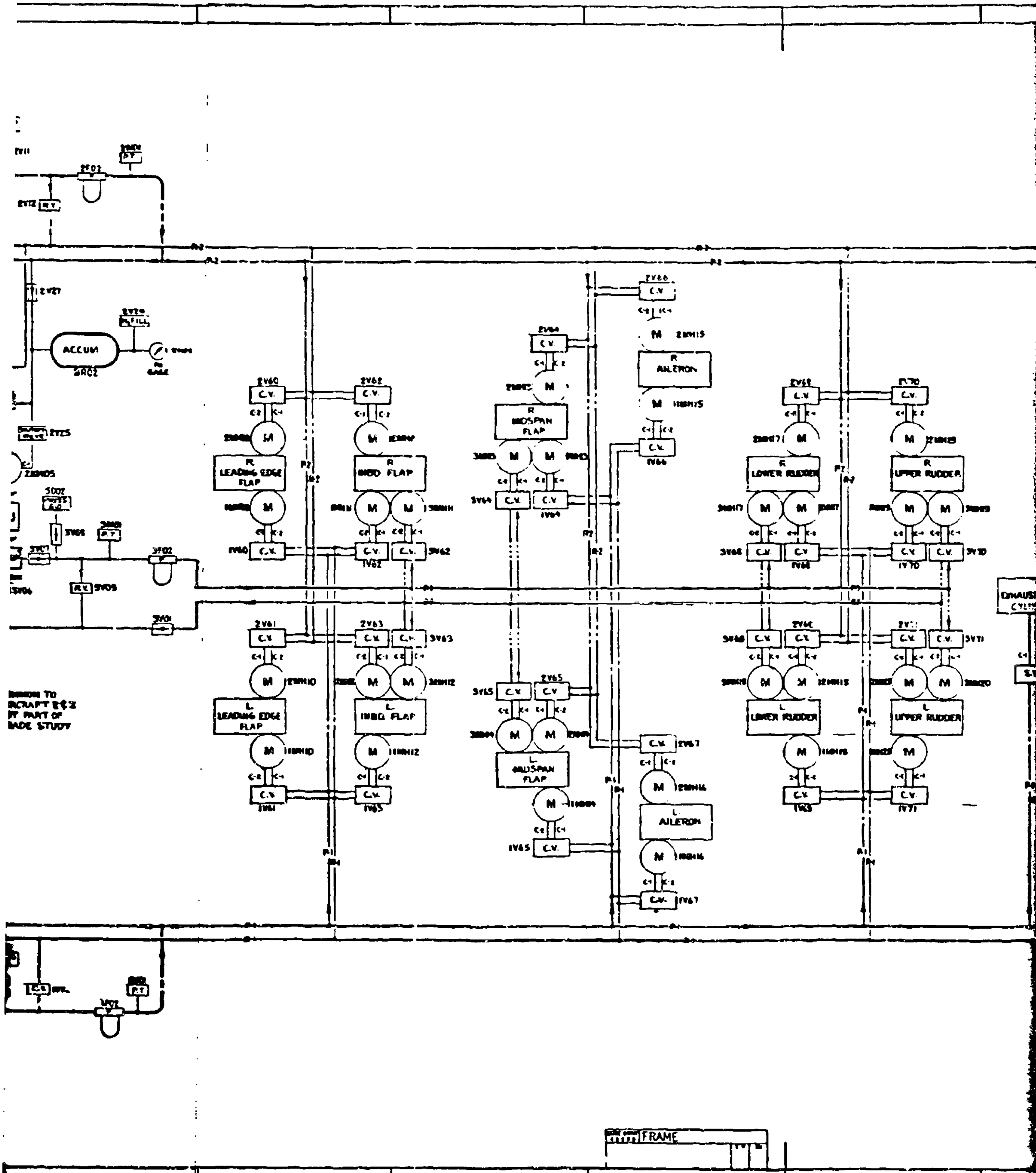

NOSE GEAR CYLINDER
NOSE GEAR CYL UP LOCK
NOSE GEAR CYL DOWN LOCK
N.G. S.Y.
N.G DOOR S.Y.
NOSE GEAR DOOR CYLINDER
NOSE GEAR DOOR LOCK CYL
R.H CANARD CYLINDER
L.H CANARD CYLINDER
C.V.
2V28
2V29
1V29
2V34
1V34
1V27
2V38
1V38
2V39
1V33
2V41
2V42
2V52
2V53
2V55
2V43
2V45
2V57
EMER DUMP

FRAME



#2 RESERVOIR
BYPASS
PUMP
R. MAIN GEAR DOOR CYLINDER
DOOR LOCK CYLINDER
R MAIN GEAR EXT./RETR CYLINDER
UP LOCK CYLINDER
DOWN LOCK CYLINDER
BRAKE RESERVOIR
BRAKE SUBSYSTEM
RIGHT WHEEL
LEFT WHEEL
ACCUM
APU
#3 RESERVOIR
EPU/APU SYSTEM
COMMON TO AIRCRAFT
NOT PART OF TRADE STUDY
L MAIN GEAR EXT/RETR CYLINDER
L MAIN GEAR DOOR CYLINDER
#1 RESERVOIR
BYPASS H/EX

3



ACCUM
R LEADING EDGE FLAP
R INBD FLAP
R MIDSPAN FLAP
R AILERON
R LOWER RUDDER
R UPPER RUDDER
L LEADING EDGE FLAP
L INBD FLAP
L MIDSPAN FLAP
L AILERON
L LOWER RUDDER
L UPPER RUDDER
C.V.
M
2V60
2V62
2V64
2V66
1V60
3V62
3V64
1V64
1V66
1V70
2V61
2V63
3V63
2V65
1V67
1V65
1V61
1V71
FRAME

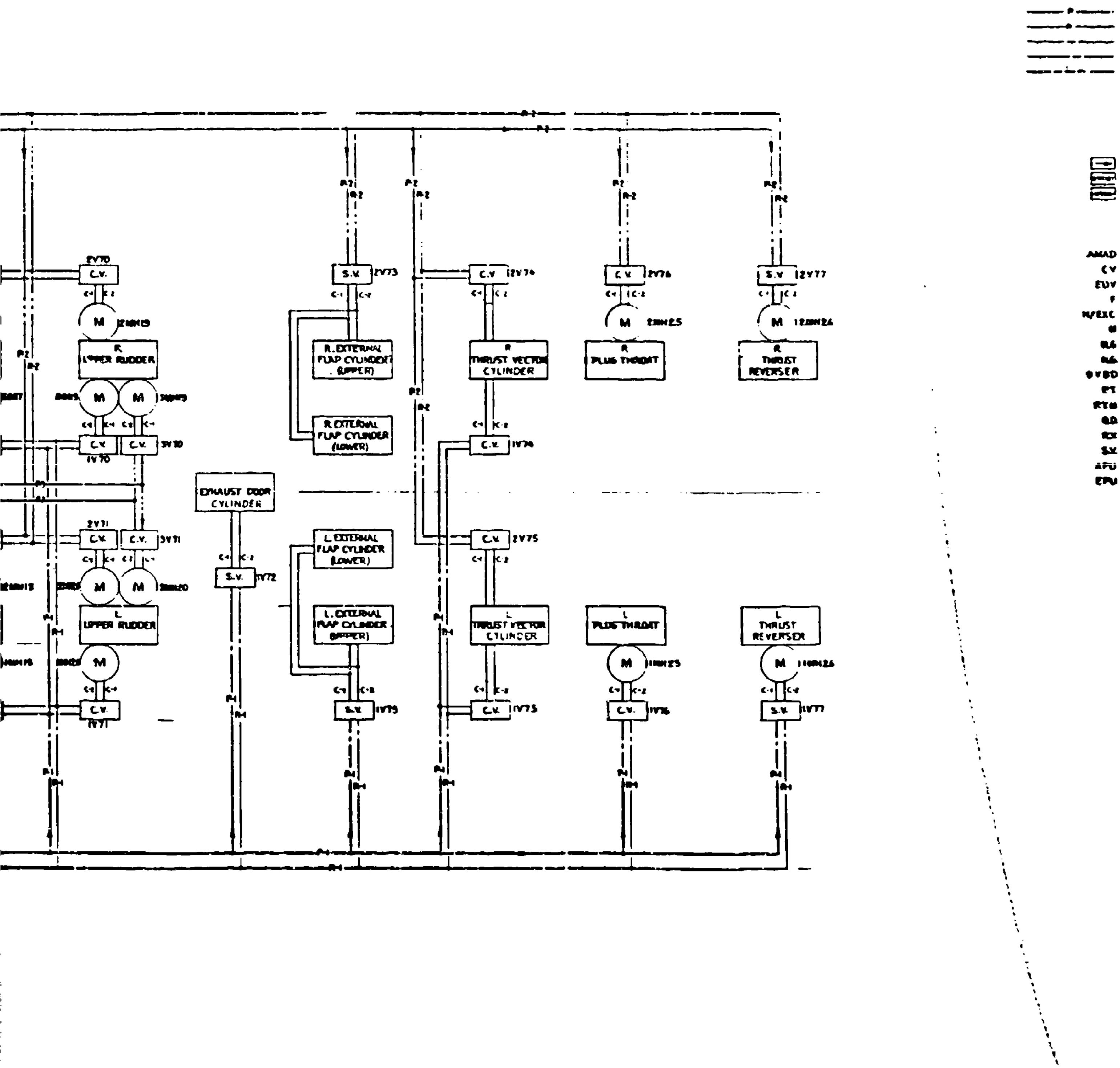


Figure 45. Schema

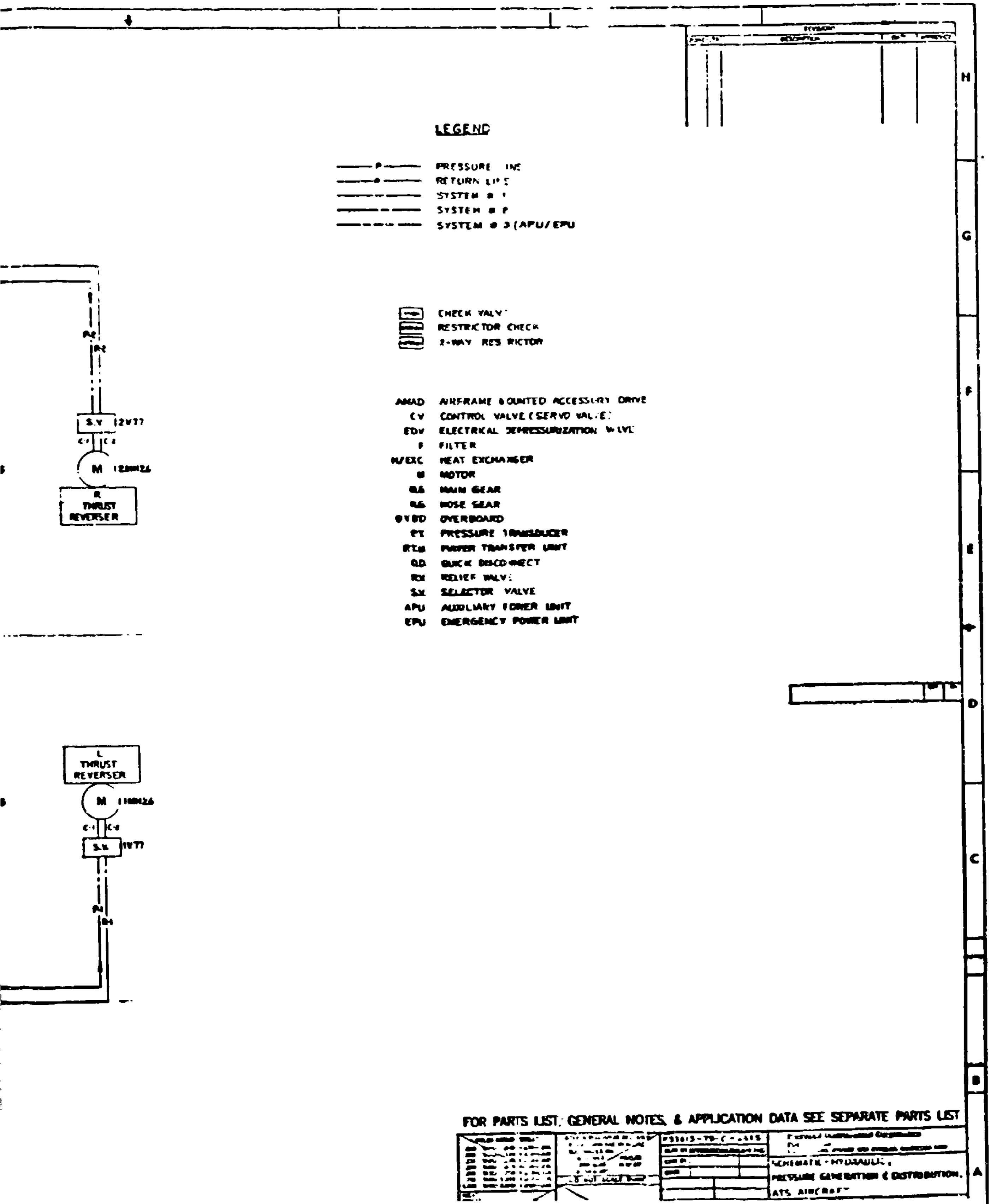
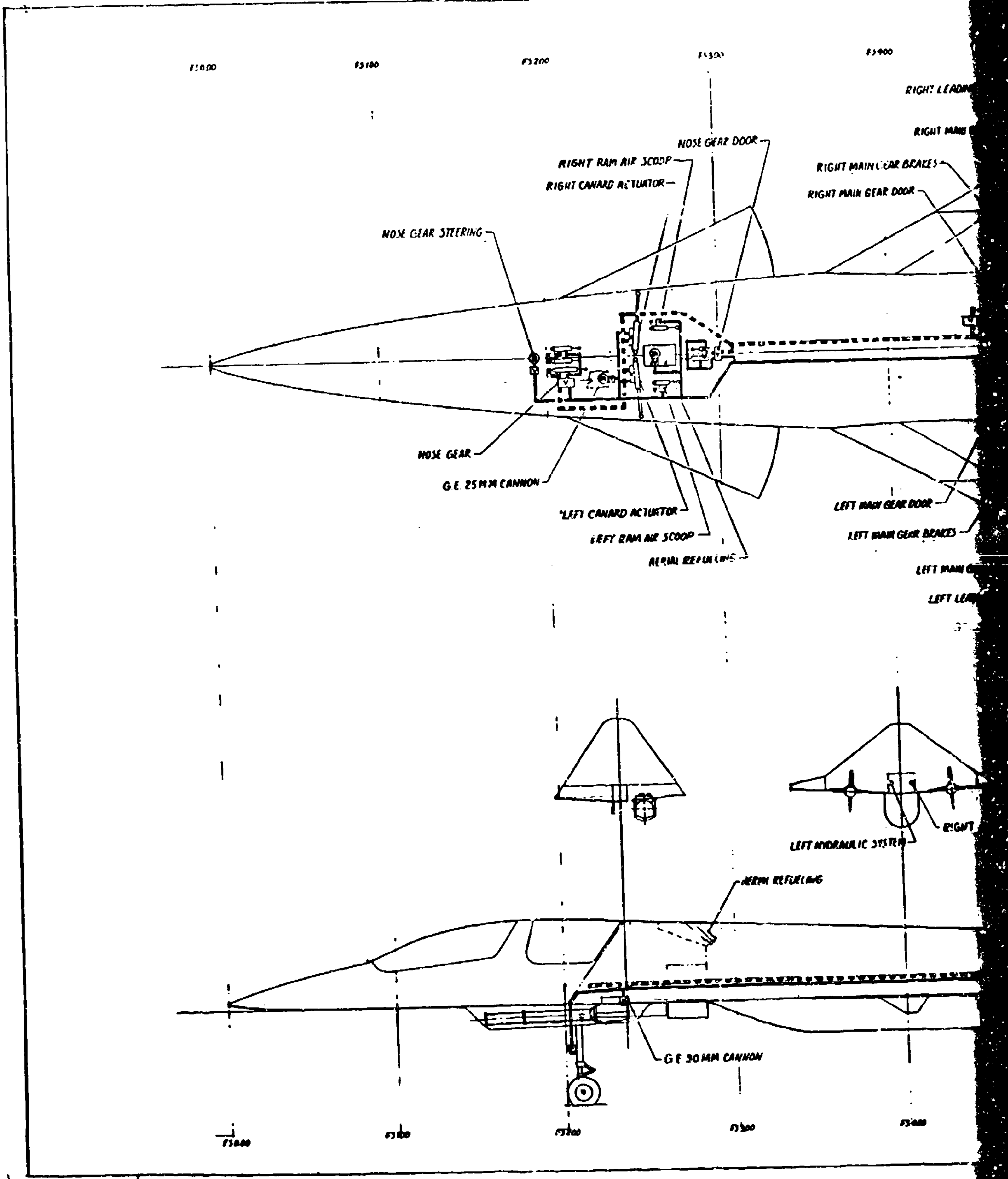


Figure 45. Schematic, Hydraulic Pressure Generation & Distribution ATS Aircraft



FS100
FS200
FS300
FS400
NOSE GEAR DOOR
RIGHT RAM AIR SCOOP
RIGHT CANARD ACTUATOR
RIGHT MAIN GEAR BRAKES
RIGHT MAIN GEAR DOOR
NOSE GEAR STEERING
NOSE GEAR
G.E. 25 MM CANNON
LEFT CANARD ACTUATOR
LEFT RAM AIR SCOOP
AERIAL REFUELING
LEFT MAIN GEAR DOOR
LEFT MAIN GEAR BRAKES
LEFT HYDRAULIC SYSTEM
AERIAL REFUELING
G.E. 30 MM CANNON
FS100
FS200
FS300
FS400

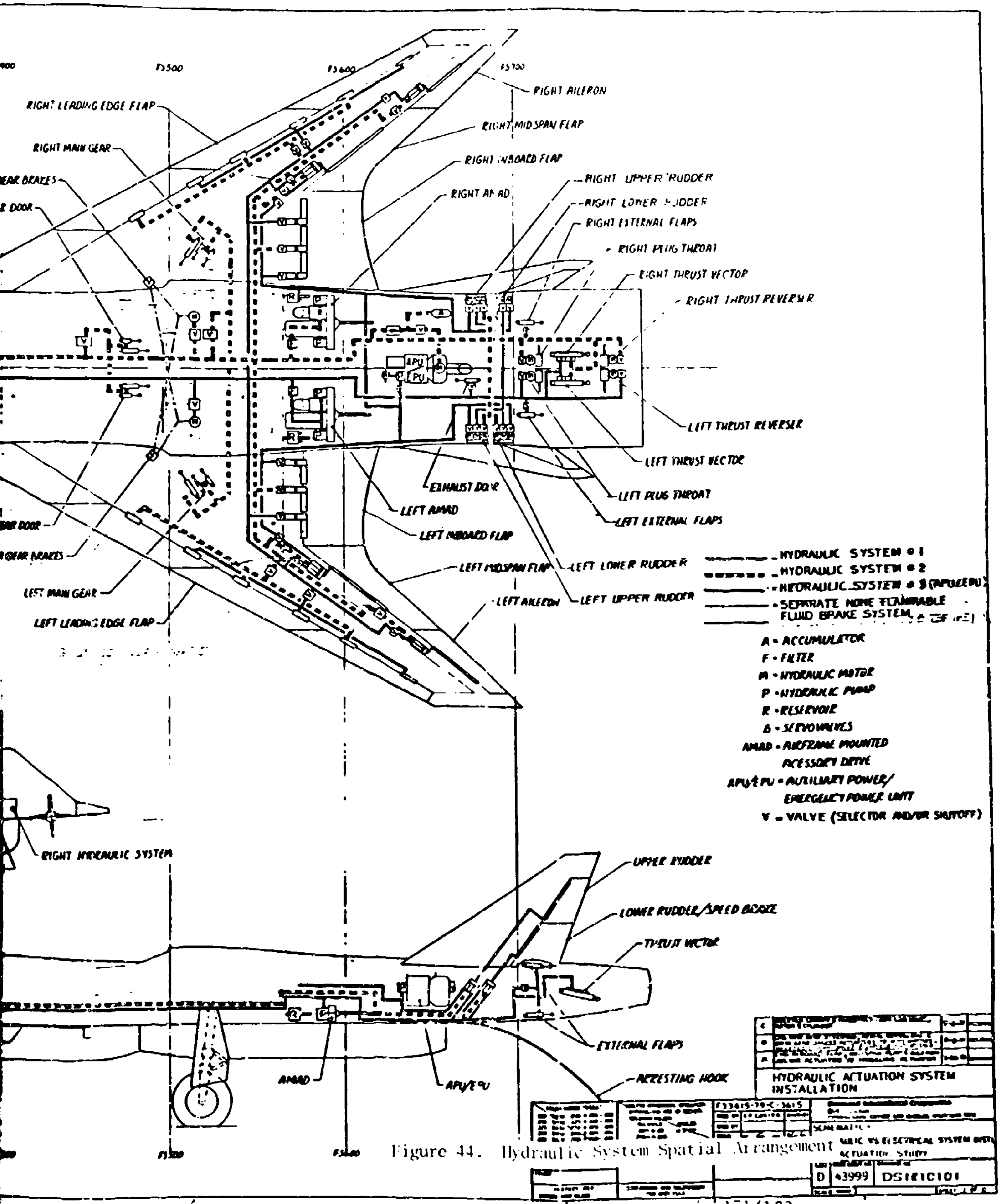


Figure 44. Hydraulic System Spatial Arrangement

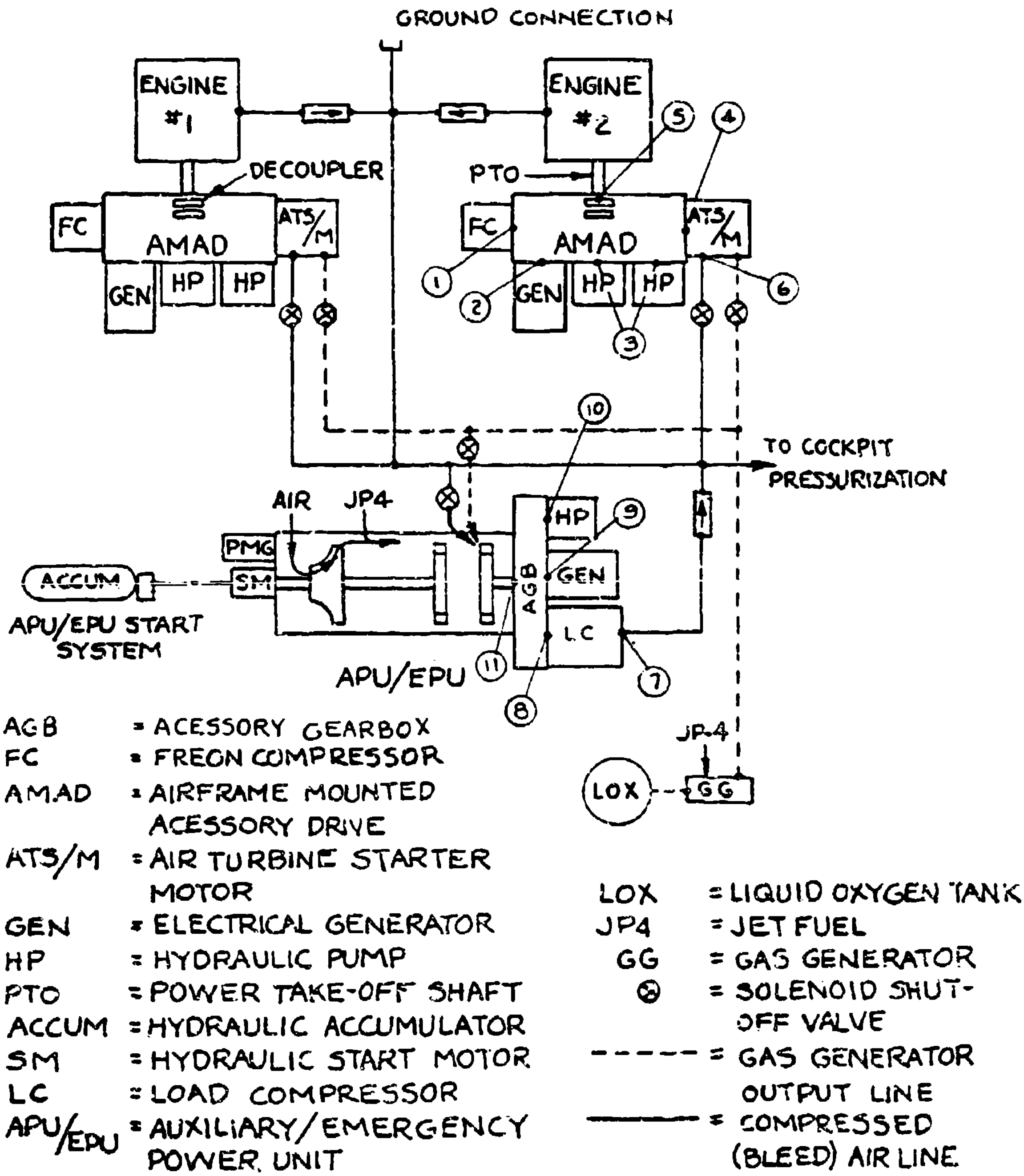


Figure 45. Aircraft II Secondary Power Generation System

compressors are limited in capability to starting one engine at a time. The ATS/M, working through the accessory gearbox and power take off (PTO) shaft, accelerates up to self sustaining speed. Start time to ground idle is nominally 35 seconds.

2. Normal Start Using Ground Cart

For engine starts using a ground cart the APU/EPU is normally not operating. Ground cart high pressure air is supplied at the ground connection; (see Figure 45) from this point the starting sequence is like B.1 above.

3. Accelerated Start

If rapid response requirements dictate, simultaneous engine starts may be achieved in either of two ways. The first is the most rapid but the least fuel efficient. In this case the APU portion of the APU/EPU is started in the normal way, after which one of the engines is started. Simultaneously the EPU gas generator is started, using LOX and jet fuel, and the second engine is started from this source. Using this approach the LOX required for high altitude engine start is largely used up and, assuming the reason for needing simultaneous engine starts is the need to "scramble" rapidly, there is no time to refuel with LOX prior to take off.

The second method for achieving simultaneous engine starts is to start pneumatically from a ground source. In this instance there is adequate power to start both engines simultaneously without the need to start up the APU/EPU. Under some circumstances this method can be faster than method No. 1 above. If, for example, air is drawn from a centralized air base air supply and is immediately available the time necessary to start up the APU and EPU in method No. 1 is avoided. The only time required is the time necessary to connect and disconnect the air hoses and to open and close the air supply valve. In those instances where the aircraft is held in alert status, lined up with the runway, with the air hose already connected, and has a tear away type of disconnect at its hose to fuselage connection, the time to start engines and to break ground is considerably reduced over that required for method No. 1.

The normal method for starting engines, however, will be to start each engine sequentially with the APU. The second engine will be either started at the ramp or while the aircraft is taxiing out to take off position.

C. APU/EPU Operation - Low Altitude

Below 20,000 ft (6096 M) altitude there is sufficient atmospheric oxygen so that the APU portion of the APU/EPU can be started and will operate continuously meeting its load demands. In this regime the

APU will supply duty cycled loads up to 70.3 H.P. for the system No. 3 (emergency) hydraulic pump and 9.8 H.P. for the emergency generator while delivering sufficient air to air start an engine. In the event that an engine cannot be restarted the APU will deliver sufficient pneumatic power to the associated ATS/M so that, with the PTO shaft uncoupled (see Decoupler Figure 45) , the hydraulic pump and generator on that gearbox (AMAD) can be driven at duty cycled power levels up to 101 H.P.. This same power level or more is available for driving either of the two AMADS on the ground for use in maintainance and checkout operations.

D. APU/EPU Operation - High Altitude

Above approximately 20,000 ft. (6096 M) altitude there is insufficient oxygen to start or maintain operation of the air breathing APU. If any emergency (such as a two engine flameout) occurs above this altitude which requires power the EPU will be turned on. Since the EPU burns LOX-JP4 (both stored on the aircraft) it can be started at any altitude. The LOX-JP4 mixture is burned in a catalytic combustor (hence nearly instantaneous light off) and the gaseous products of combustion are directed to the APU's free turbine. The free turbine, through the APU's output gear train, drives the emergency pump and generator, however, the load compressor is unloaded. The emergency pump and generator supply the electrical and hydraulic power necessary to fly, or glide, the aircraft down to an altitude at which the APU can be started and engine start attempts can be made.

4.2.1.2 Brake System - It was decided that the brake system would remain hydraulic for both versions of the study aircraft (hydraulic and electrical). In each instance the brake system would use the newly developed chlorotrifluorethylene type non flammable hydraulic fluid and would be a small separate system independent of the main hydraulic or electrical power generation and distribution systems. This decision was made based upon consideration of several factors:

A. If aircraft of the mid 1990's retain hydraulics as a prime element of power generation and distribution in their secondary power systems, they would almost certainly use a separate non-flammable fluid subsystem for their braking system. This was felt to be true because of the many aircraft losses which have been traceable to brake fires fed by the currently used flammable hydraulic fluids.

B. Brake systems for future high performance military aircraft would have an advanced version of a fly-by-wire type of antiskid. In this approach the pilot's brake input and the incipient skid sens would both be transmitted as an electrical signal and would be mixed electronically to provide a modified output signal. This output signal would be sent to an amplifier in the form of a metering valve for the hydraulic system or an inverter for the electrical system. The four servo controlled metering valves required for the hydraulic approach would weigh approximately 2 lbs while the 4 ambient air cooled inverters for the electrical approach will weigh at least 12 lbs.

C. Electrical brake systems have been under development for several years, however all work to date, has been analytical; no hardware has been built and no components have been tested. Based on the analysis and some optimism it was felt that electric brakes might be developed to the point where they were nearly equivalent in weight and reliability to current hydraulic brakes. There was, however, continuing doubt that they would meet the wet and dry runway anti-skid performance requirements currently met by hydraulics.

The electric brake's primary item of desireability was the fact that it would eliminate brake fires. A secondary advantage was the fact that electric brakes offered the possibility of reduced routine maintenance requirements relative to a hydraulic system and particularly to a separate hydraulic system (actually two small separate hydraulic systems with two separate reservoirs) using non-flammable fluid, such as planned for the ATS aircraft.

Considering items A through C it was apparent that the scales were tipped in favor of the hydraulic approach. The fact that the prime virtue of the electric brake (no brake fires) was offset by the use of non-flammable fluid in the hydraulic approach was instrumental in shifting the balance radically. In effect, based on what was known at the time of this report, reasonable extrapolation of the state of the art to the mid 1990's would still favor the hydraulic approach in the following areas:

Weight
Steady State (stalled) Power Demand
Heat Rejection During Braking
Reliability
Unscheduled Maintenance
Performance (anti-skid capability)

Only in the area of scheduled maintenance would the electric brakes have had a clear superiority. Because the non-flammable fluid brake subsystem could have been in both baseline aircraft without impairing the "all electric" power generation and distribution characteristics of the aircraft, it was decided that this was the approach to use.

As previously indicated, the brake system became an arrangement consisting primarily of two small compact hydraulic systems employing non-flammable hydraulic fluid; each having its own pump, reservoir, filters, control valves and actuators. Each hydraulic system powered brakes on both the right and left hand main gear wheel with one system acting as a backup for the other. One system had an accumulator which could supply limited emergency braking in addition to its basic function of providing parking brake capability. The pressure compensated variable delivery pump for each hydraulic system was driven by a hydraulic motor on the hydraulic aircraft (Aircraft II) and by an electric motor on the all electric aircraft (Aircraft I). A schematic showing the brake system is included as part of Figure 43.

4.2.1.3 In-Flight Refueling - It was decided that the standard (UARRSI) inflight refuel receptacle would be used in both vehicles. This decision was based on Rockwell experience in attempting to use a non standard refuel receptacle in the B-1 aircraft. Even though the change was modest; substituting 4000 psi actuators, valves, and plumbing for the standard receptacle's 3000 psi components and saving a little weight in the process, it was not bought. The Air Force felt that it was of such overriding importance that the standard receptacle be used, and thus be warehoused and available for use interchangeably on any Air Force aircraft, that they were willing to give up the potential weight saving and suffer an additional weight penalty to avoid the use of a non-standard receptacle. The additional weight penalty was that represented by the pressure reducers and pressure relief equipment which was necessary to adapt the B-1's 4000 psi system to the 3000 psi receptacle. Since the standard unit is even more widely used now than it was at the time of the B-1 decision, and considering that its use will be even more extensive during the 1990's, it was felt a standard receptacle was nearly mandatory unless some truly compelling reason legislated to the contrary. There appeared to be no compelling reason.

Both the electric and hydraulic baseline aircraft could provide 3000 psi hydraulic power for the standard receptacle thru the use of power transfer unit (PTU). Essentially a PTU was a specialized version of a motor pump. Such a device (PTU) was mandatory for the "all electric" baseline aircraft if a standard (UARRSI) receptacle was to be used. Theoretically a pressure reducer (rather than a PTU) could have been used in the 8000 psi hydraulic baseline aircraft to adapt to the standard receptacle. However, return line pressures near the receptacle in an 8000 psi system would often exceed 3000 psi. Since a pressure reducer would have been referenced to return pressure near the receptacle, the pressure in the receptacle components would always be higher than return pressure and often much higher than 3000 psi. These high pressure return conditions could be largely avoided by running a dedicated return line 30 ft back to the reservoir. However, this would involve an added weight penalty of 4 lbs and still leave a serious doubt as to whether the allowable return pressures would not intermittently be exceeded at low temperatures. For these reasons use of a pressure reducer for this application was considered unacceptable. Therefore, as indicated above, a PTU was used for the hydraulic baseline as well as for the electric. In the hydraulic baseline case the PTU consisted of a 3000 psi pressure compensated pump supplying the receptacle driven by an 8000 psi constant displacement hydraulic motor. In the case of the "all electric" airplane, the hydraulic motor would be replaced by an electric motor but the 3000 psi pump would remain unchanged. In affect everything downstream (on the receptacle side) of the hydraulic pump mounting flange (interface point) was identical for both the hydraulic and electrical baselines. The "downstream" items were the pump (1.9 lb), a reservoir module (3.8 lb) containing pressure, case drain and return filters, relief valves, fluid level gages etc., plumbing and fittings (0.3 lb), and the receptacle itself.

Under these circumstances an interface was created which, once its transmitted power and rotational speed were defined, could be driven either by hydraulic or electric motors. The power at the interface and the rotational speed is given in Table 9.

4.2.1.4 Landing Gear - The nose gear was extended and retracted by both main hydraulic systems (See Figure 42) hence a tandem linear actuator was used. The main gear was retracted and extended by system #2 only. Upon loss of system #2, emergency extension was obtained by free fall, with the weight of the gear the prime mover. Conventional up and down locks, and fairing door actuation was employed. Run-around valving allow for gear operation with the aircraft on blocks.

4.2.1.5 Pump Sizing - The peak hydraulic system flow demands, and the ones which sized the system, were those which occured during combat. The magnitude of the flow demands were derived from the loads shown in Tables 9 and 12 and were tabulated in Table 29 to show the loads in each system which apply during combat. In a great many aircraft the peak flow demand, in relation to pump capacity, occurred during landing flare-out when flight control and landing gear demands were high and the pump capacity was low because the engine was at idle RPM. This was not true of this aircraft, however, since the flow demands of gun, thrust reverser, and plug throat operations during combat far exceeded flare-out demand. Therefore, the pumps for all three systems were sized based on the total flow demand load (in KW) shown in Table 29.

The maximum flow rating of the pump (in GPM) was determined using the following formulas and assumptions:

Formulas:

$$\text{Flow (GPM)} = \frac{\text{Power (HP)} \times 1714}{\text{Pressure (PSID)}}$$

and

$$\text{Power (HP)} = \frac{\text{Power (KW)}}{.746}$$

Assumptions:

The actual maximum power demand was 2/3 of the the theoretical (summed) power demand, since in a group of actuators such as those listed in Table 29, not all actuators would be operating at a given time, and of those operating, not all would be operating at their peak demand capability. This was a restatement of a basic "ground rule" already given in paragraph 2.4.4.

The effective pressure at the load was 4667 PSID in an 8000 PSI system based on the following:

	8000 PSID	System pressure
Less	500 PSID	Lost in supply line
Less	500 PSID	Lost in return line
Less	2333 PSID	Lost in valving (1/3 X 7000 PSID)
	4667 PSID	Net at operating load and rate

TABLE 29. AIRCRAFT II LOADS DURING COMBAT (5 SEC. LOADS)

• POWERED BY SYSTEM
— NOT POWERED BY SYSTEM

N/A NOT OPERATED DURING THIS MISSION PHASE

FUNCTION		SYSTEM #1	SYSTEM #2	SYSTEM #3	LOAD (KW)/ACT. AT INTERFACE (1)	LOAD (KW)/ACT. AT ACT. PORTS	NO. OF ACT/SYS	LOAD/SYS (KW) #1	LOAD/SYS (KW) #2	LOAD/SYS (KW) #3
INBOARD FLAP		•	•	•	14.87	17.49	2	35.0	35.0	35.0
MIDSPAN FLAP		•	•	•	3.02	3.55	2	7.1	7.1	7.1
AILERON		•	•	—	1.11	1.30	2	2.6	2.6	—
UPPER RUDDER		•	•	•	0.55	0.65	2	1.3	1.3	1.3
LOWER RUDDER		•	•	•	0.52	0.61	2	1.2	1.2	1.2
LEADING EDGE FLAP		•	•	—	1.77	2.08	6	12.5	12.5	—
CANARD	LEFT	•	•	—	0.23	0.27	2	0.2	0.3	—
	RIGHT	•	•	—				0.3	0.2	
THRUST VECTOR VANE	LEFT	•	•	—	0.72	0.72	2	0.3	0.4	—
	RIGHT	•	•	—				0.4	0.3	—
EXTERNAL FLAP	LEFT	•	—	—	4.13	4.13	2	8.3	—	—
	RIGHT	—	•	—				—	8.3	—
PLUG THROAT	LEFT	•	—	—	28.67	33.73	1	33.7	—	—
	RIGHT	—	•	—				—	33.7	—
THRUST REVERSER	LEFT	•	—	—	22.84	26.87	1	26.9	—	—
	RIGHT	—	•	—				—	26.9	—
NOSE GEAR ACTUATION		•	•	—	1.40	1.40	1	N/A	N/A	—
MAIN GEAR ACTUATION		—	•	—	2.71	2.71	2	—	N/A	—
NOSE GEAR STEERING		•	—	—	0.48	0.56	1	N/A	—	—
MAIN GEAR BRAKES		•	•	—	3.18	3.74	2	N/A	N/A	—
RIGHT RAM AIR SCOOP		•	—	—	0.93	0.93	1	N/A	—	—
LEFT RAM AIR SCOOP		•	—	—	0.01	0.01	1	N/A	—	—
EXHAUST DOOR EQUIPMENT BAY		•	—	—	0.01	0.01	1	N/A	—	—
ARMAMENT		•	—	—	35.05	41.24	1	41.2	—	—
REFUELING RECEPTACLE		•	—	—	0.83	0.98	1	N/A	—	—
TOTAL								171.0	129.7	44.6

(1) LOADS FROM TABLES 9 AND 12

Using the preceding formulas and assumptions the required flow from the pump at military (combat) power became:

$$\text{Flow (GPM)} = \frac{2 \text{ Power (KW)} \qquad 1714}{3 \times .746 \quad (\text{Pressure PSID})}$$

$$= \frac{2 \times (\text{KW}) \times 1714}{3 \times .746 \times 4667} = .328 \times (\text{KW})$$

Therefore the required flow for the three systems using the load demands from Table 9 was as follows:

(System No. 1)	.328 X 171.0	= 56.09 GPM
(System No. 2)	.328 X 129.7	= 42.54 GPM
(System No. 3)	.328 X 44.6	= 14.6 X 3/2 = 21.9 GPM

The 3/2 factor was added to system 3's determination of required flow to cancel out the effect of the 2/3 simultaneous flow assumption. This was done because the emergency system (i.e. system 3), unlike the primary systems, had relatively few actuators all of which were very likely to be working at maximum power simultaneously during an emergency.

The foregoing statement might seem to be in conflict with the approach used in handling emergency loads in aircraft I however it actually was not. Electrical power systems differ from hydraulic in at least two vital areas as follows:

1. Electrical system load demands tend to follow and be proportional to the torque demands of its actuators, whereas, hydraulic system load demands tend to follow and be proportional to the output velocity or rate of its actuators. On this basis, as pointed out and discussed in paragraph 4.1.5.1.1, electrical systems would exhibit a peak load demand on the generating system at stall, whereas hydraulic systems would impose their peak demand at maximum surface rate-no load conditions.

2. Aircraft hydraulic power generating systems are power limited. Once the maximum displacement of the pump (or pump) is achieved at a given speed an absolute limit in power generation capability is reached which cannot be exceeded. Electrical power generation systems, in contrast, can exceed their continuous power rating by 250% for short periods.

Based on these differences it was felt that the 2/3 rule should be suspended when considering emergency conditions and hydraulic power systems in this type of aircraft. Although it was highly improbable that an aircraft would find more than one of its major emergency control surfaces approaching stall while 2/3 of its remaining surfaces were at design load and rate when recovering from a maneuver after a combat induced emergency, it was highly probable that under the same conditions all emergency surfaces, even though experiencing low aerodynamic loading, could be asked to move simultaneously at maximum rate for short periods of time. Since the hydraulic system was load limited in terms of rate and thus did not have the forgiveness of an electrical system, it was decided that all emergency control surfaces should be able to meet their maximum rate requirements simultaneously.

Based on the preceding data a pump size of 28 GPM was selected. This meant that two pumps would be required for systems No. 1 and No. 2 while one would be more than adequate for system No. 3. It could be seen that, with this size selection, pump capacity for systems No. 2 and No. 3 was oversize (13.46 GPM for system No. 2 and 6.1 GPM for system No. 3). Pumps could have been selected which were exactly sized for each of the three systems, however, this would have tripled the number of pump types in the logistic pipeline. The negative impact on life cycle [illegible] arising from the increased maintenance and stocking problems, would more than offset any positive impact from the exactly sized pumps reduced weight.

In addition to the standardization advantages resulting from using five uniformly sized pumps, and assuming one exactly sized pump for each system as the alternate, the standardized pump approach allowed more installation flexibility in system No. 1 and No. 2 in that their smaller size made it easier to tuck the pump into available spaces without the danger of bumping the mold line or causing major structural revisions. The individual pump units weighed less than half as much as their larger sized alternates, therefore, they were much easier to handle during maintenance operations. It should be remembered also that, even though the selected pumps for systems No. 2 and No. 3 were oversized, the plumbing systems were sized for the actual design flow and, not the rated flow of the pumps. For this reason the plumbing system remains unchanged and does not grow as would be expected if it were sized to meet pump ratings. Therefore, the weight chargeable to the use of multiple standardized, but oversized, pumps was nearly negligible.

4.2.1.6 Actuator Installation Design - It was recognized that the lightest weight and most efficient actuator design for most of the actuation functions on the aircraft would be a conventional piston type linear actuator if it could be made to fit inside the aircraft mold line. Although widely used for most actuation functions in the past, this approach had encountered difficulties, when applied to the flight control functions of more modern aircraft, because of the higher loads and thinner wings, characteristic of these aircraft. The wings on the ATS aircraft tended to be even thinner and the loads higher. However, in spite of this fact, it was felt that, because of the potential actuator size reduction and load output increase resulting from the use of an 8000 psi system, a conventional linear actuator might still be useable. To verify this a study was made to determine the viability of attempting to install a linear actuator to perform the aileron actuation function. This function was selected because it represented one of the tightest installation envelopes (See Figure 13 and paragraph 2.3.1.6) on the aircraft.

After examining several approaches, such as remote located actuators working through a series of bellcranks to power the surface, an approach involving direct actuation was settled upon. This approach is shown in Figure 46 and 47. In order to fit the actuator into the wing it was necessary to make each actuator as flat as possible. To do this each actuator was designed to consist of three balanced pistons in parallel. This helped to reduce the rod and piston sizes to a point where a flat pancake like housing could be created which would fit inside the wing mold line between the rear spar and hingeline.

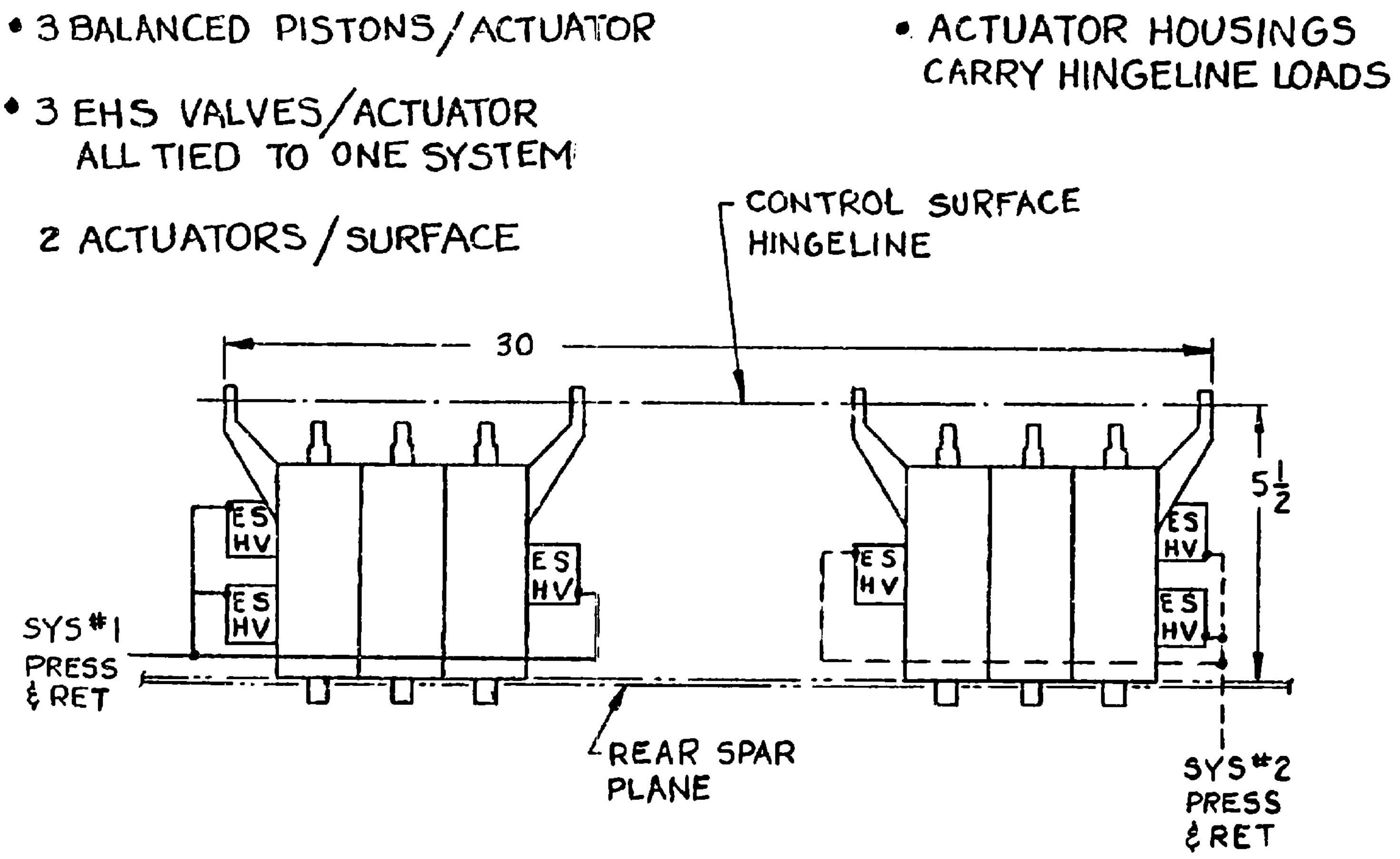


Figure 46. Aileron Actuator Instl.

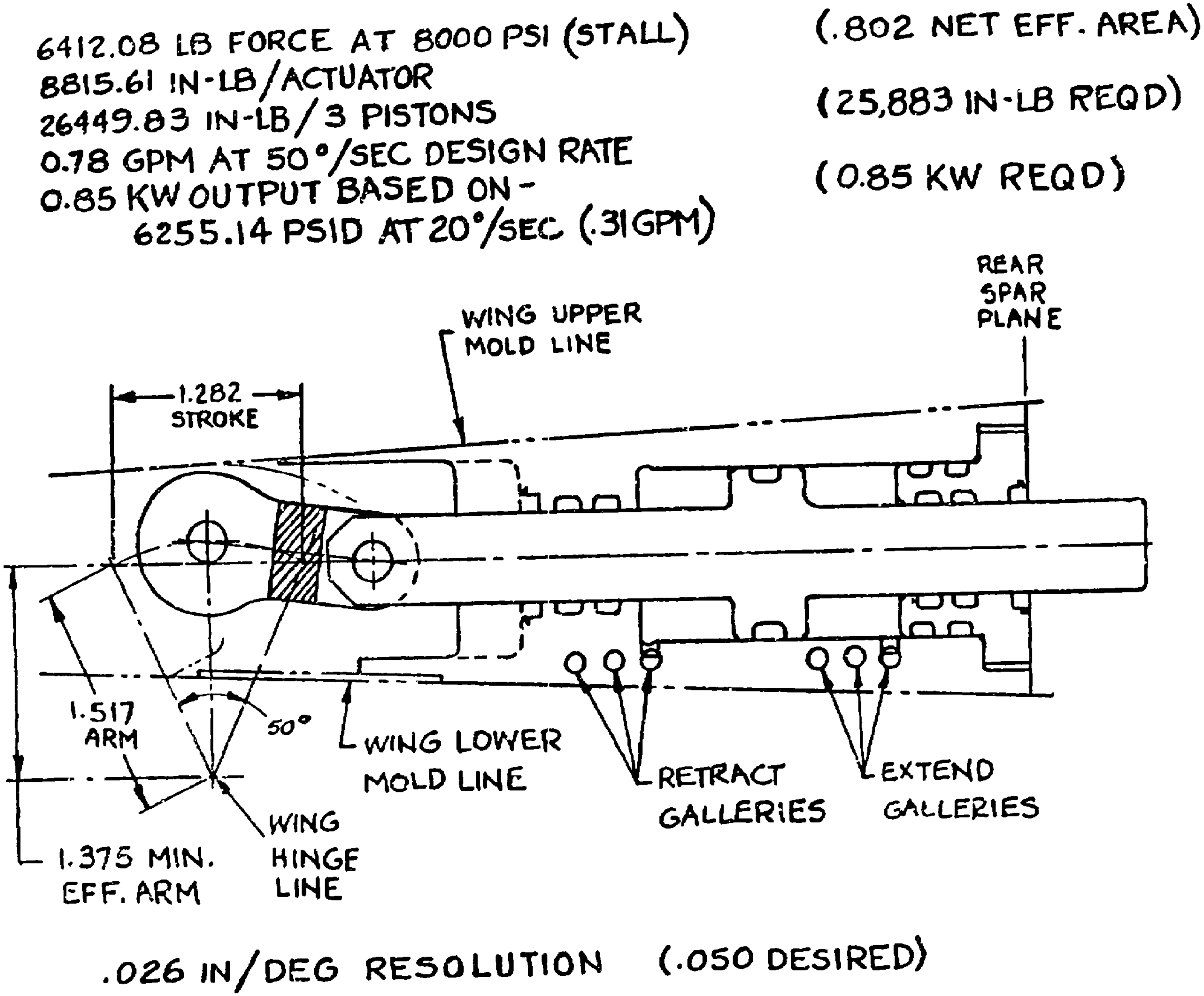


Figure 47. Aileron Actuator

As can be seen in Figure 46 three direct acting (single stage) electro-hydraulic servo valves (EHSV) were used for each actuator and two actuators were used for each surface. The reason for the use of 3 EHS valves was two fold; first, a single large direct acting valve, capable of handling the flow requirements of the three parallel pistons in a given actuator, did not fit within the mold line of the wing whereas, three smaller valves would fit; and second, the use of three valves was very compatible with the 4 channel (three control channels plus model channel) fly-by-wire system used on the aircraft. In order to save space and keep weight to a minimum, the actuator housing was designed to transfer the control surface reaction loads generated by the hingeline to the rear spar.

Figure 47 shows a crossection view of one of the actuators. It shows that, in spite of the heroic measures taken to keep all elements of the actuator within the wing mold line and within the fore and aft envelope, it none-the-less violated both. One end of the balanced piston penetrated the rear spar, which was undesireable and constituted a weight penalty, and the hingeline was displaced 5/8 inch below the lower wing mold line. This meant that at least 4 fairings 1-1/2 inches deep by 6 inches long by 1-1/2 inches wide would have been required to cover the hinge mechanism. Even though the drag induced by these fairings might have been considered acceptable, the design was unacceptable for other reasons. The performance data shown on Figure 47 shows that the actuator would meet hinge moment and rate goals, however, it shows that the actuator was deficient in resolution. In order to meet the .050 in/deg resolution requirement the hingeline pivot arm would have had to have been increased from the 1.517 inches shown to 2.92 inches. This would have doubled the size of the fairings, doubled the stroke and, through doubling the stroke, would have increased the length of the actuator by several inches thus wiping out the rear spar. This was considered entirely unacceptable. After failure to meet the aileron envelope and performance requirements this same general approach was cursorily examined for application to the midspan and inboard flaps. Here too it could not meet requirements and further consideration of this approach was dropped.

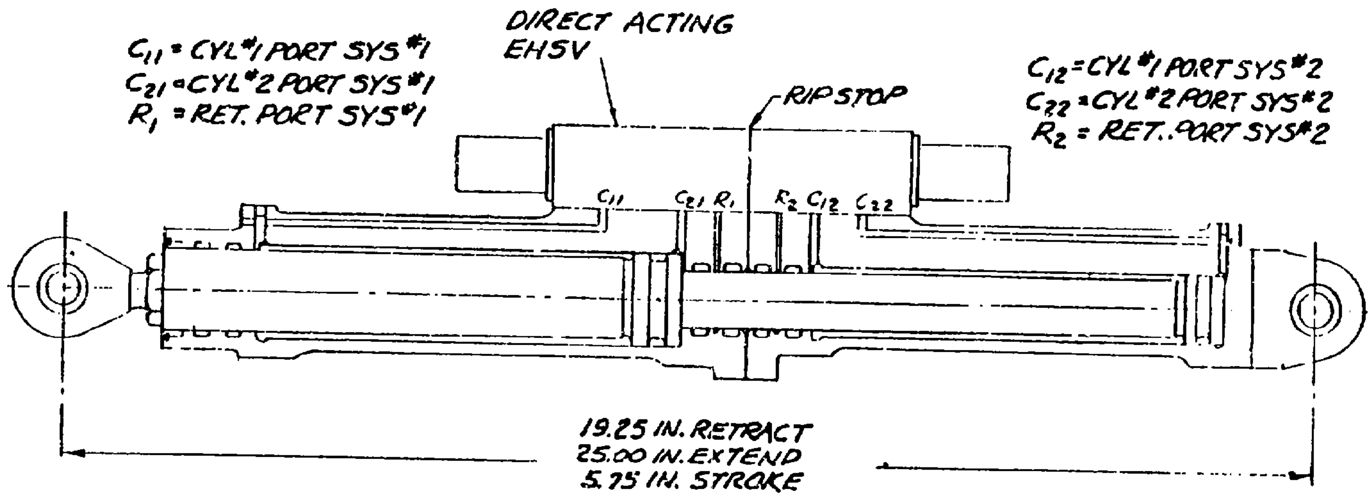
At or about the time this decision was being reached the data discussed in paragraph 4.1.5.1 was received from Airesearch and it became apparent that the best possible solution for most of the control surfaces was to use a mechanical power hinge and drive it with a motor (electrical motor for the electrical system and hydraulic motor for the hydraulic system). From this was formulated the general ground rule (see paragraph 2.4.5) which was made a basic part of the program, and which stated in effect that, where the hydraulic system (aircraft II) uses motors, the electrical system (aircraft I) shall use motors, and vice versa, and also stated that, where the hydraulic system uses linear actuators, the electrical system shall use the electrical verision of a linear actuator- the ballscrew.

The ground rule of paragraph 2.4.5 evolved gradually as the program progressed. Epicyclic geartrain (power hinge type) actuators were looked at as a possible means for actuating the landing gear and landing gear doors on aircraft I. However, after extensive consideration of the difference in load paths between a power hinge, which reacts its loads in the immediate area of the trunion or hinge line, and a typical linear hydraulic actuator, which reacts its loads

to remote structure, it was decided that the errors in assessing weight impact which could arise from incorrect assumptions about structural load paths, and the nature of the structure to which either type of actuator attached, could far outweigh any inaccuracies or unfair weight penalties which arose from adopting the ground rules of paragraph 2.4.5. Examination of other actuation functions, such as the canard lead to the same general conclusion and when it became apparent that 11 other flight control functions would be power hinge operated whether, hydraulic or electric, the ground rules of paragraph 2.4.5 were adopted.

As indicated above the canard actuator was examined in some detail since it was considered representative of those applications where a linear hydraulic actuator could be used advantageously. Figure 48 shows the canard actuator as it finally evolved. It can be seen immediately that the stroke was reduced from that shown in Table 8 and also in Table 12. The Table 8 stroke was 15.36 in. and that given in Figure 48 for the final hydraulic system actuator design, was 5.75 in. or slightly more than 1/3 of the original requirement. The reason for this change was two fold; first the 24 in. installed (retracted) length assigned to this actuation function on the aircraft would not accommodate a tandem cylinder design employing a stroke of 15.36 in. and second, even if it had, the resulting cylinder (particularly in an 8000 psi configuration) would have been so long and slender that it would have been unstable as a column in compression. Therefore the actuator was relocated to a point closer to the pivot point of the canard surface such that a 5.75 in. stroke with a 6400 lb stall capability and a 5,333 lb design operating load capability would meet the canard surface's requirements. The output design power requirements of course, remained unchanged at 0.17 KW. Although the electric version of the actuator (see Figure 31) was not restricted by the tandem actuator requirement of the hydraulic version and, therefore, could meet the original retracted length and stroke requirement, it too was dangerously close to column instability and could profit from meeting the new shorter installed length. Therefore, it was assumed, for the purposes of aircraft I definition, that the installed length of the ball screw actuator was 19.25 in. Instead of the 24 in. shown in Figure 31 and that, in common with it's aircraft II hydraulic counterpart, it's stroke was 5.75 in. and its stall load was 6400 lb. The weight of the ball screw unit, however, remained essentially unchanged at 15 lb since the weight increase associated with the increased load offset the weight decreased associated with decreased installed length.

It is interesting to note that the ball screw unit with its two motors and torque summing gearing plus brakes actually weighed less than the 8000 psi hydraulic actuator. This resulted from the fact that the hydraulic version of the canard actuator, shown in Figure 47, included two EHSV valves which were the power switching and control devices equivalent to an inverter in the electrical approach. When the weight of the two inverters required for the electrical approach (10 lb each per Table 21), were included the total weight of the electrical actuation function became 39 lbs versus 21.25 lbs for the hydraulic.



SYSTEM #1 SIDE

BORE = 1.375 DIA. = 1.485 IN^2 GROSS AREA
ROD = 1.250 DIA. = 1.227 IN^2 GROSS AREA

C_{11} NET AREA = 1.485 - 1.227 = .258 IN^2
C_{21} NET AREA = 1.485 - 0.601 = .884 IN^2

SWEPT VOLUME EXTEND = 5.083 IN^3
SWEPT VOLUME RETRACT = 1.484 IN^3
EXCHANGE VOLUME = 3.599 IN^3

SYSTEM #2 SIDE

BORE = 1.032 DIA. = 0.836 IN^2 GROSS AREA
ROD = 0.875 DIA. = 0.601 IN^2 GROSS AREA

C_{12} NET AREA = 0.836 - 0.601 = 0.235 IN^2
C_{22} NET AREA = 0.836 IN^2

SWEPT VOLUME EXTEND = 4.807 IN^3
SWEPT VOLUME RETRACT = 1.351 IN^3
EXCHANGE VOLUME = 3.456 IN^3

WEIGHT OF 3000 PSI ACTUATOR (FROM FIGURE 1 REFERENCE 27) = 9.6 LBS
WEIGHT CORRECTION $= 9.6 \times \frac{8000\ PSI}{3000\ PSI} \times .83 = \boxed{21.25\ LBS}$ FOR 8000 PSI ACTUATOR

Figure 48. Canard Actuator Design

4.2.1.7 Actuator Sizing - The basic characteristics of the hydraulic linear actuators used on aircraft II are listed in Table 30. All of these actuators, with the exception of the canard actuator, performed utility functions. Three of the actuators (thrust vector vane, nose gear, and canard) were tandem unbalanced, and the rest were single unbalanced. All the actuators were checked for column stability and found to have adequate margin. The retract lengths were verified as being adequate for the alternate ball screw type electromechanical actuators used on aircraft I. The 3000 psi weight was determined based upon swept volume and the data given in Reference 22. This weight represented an actuator which had 3/8 of the required power output (i.e. the ratio of 3000 psi to 8000 psi). At the same swept volume, but operating at 8000 psi and hence 100% of required power output, it was assumed that the actuator would weigh 8/3 times the 3000 psi actuator's weight times 83% giving a weight correction factor of 2.21 in going from 3000 psi to 8000 psi. The 83%, used in determining the correction factor, arose from the consistent 15 to 17% weight reduction reported in Reference 7. These two analyses represented two independent and rather detailed design analyses for substituting various 8000 psi actuators for 3000 psi actuators in two aricraft (F-14 and A-7). In each instance these actuators performed the same actuation function at the same installed length.

A plot from the actuator weight data presented in Reference 22 was made and is shown as the solid (3000 psi) lines in Figure 49. The solid linesnes represented a mean value of the scatterband of weight values presented in Reference 22, for a tandem unbalanced utility type actuator (lower solid line). The upper and lower dashed lines represent corresponding data for 8000 psi actuators of the same swept volume and were determined by applying the 2.21 correction factor, previously mentioned, to the 3000 psi data. The validity of the resulting curves was further verified by comparing them to extensive in-house data on 4000 psi equipment. In each case the weight data presented in Figure 49 represents that of the complete (dry) actuator ready for installation. For utility actuators this was considered to include typical plumbing and support brackets mounted on the actuator prior to installation and for flight control actuators it was considered to include, not only plumbing and brackets, b the direct acting electro hydraulic servo valve (EHSV) as well. The weight of an actuator filled with fluid was considered to be 110% of the dry weight for 3000 psi actuators and 102% for 8000 psi units.

4.2.1.8 Accumulator Sizing - It should be noted in Figure 45 that an APU/EPU start accumulator was required for aircraft II and it could further be seen in Figure 42 that this accumulator was plumbed into system #2 and was the only accumulator used on the aircraft. Since accumulators tend to have a "double barreled" impact on the system in which they are used by virtue of the fact that, not only is their size and weight directly additive to the system, but they have a parallel and proportional size and weight impact on the reservoir servicing that system. This arises from the fact that the reservoir must be increased in size to accept the fluid discharged from the accumulator and must be further increased to provide additional

TABLE 30. HYDRAULIC LINEAR ACTUATORS

ACTUATOR	STROKE (IN.)	RETRACT LENGTH (IN.)	DESIGN LOAD (LBS.)	ROD DIA. (IN.) / AREA (IN²)	BORE DIA. (IN.) / AREA (IN²)	NET AREA (IN²)	EXCH. VOL. (IN³)	SWEPT VOL. (IN³)	3000 PSI ACT. ① WT. (LB)	CORR. FACTOR ②	8000 PSI ACT. WT. (LB)
EXTERNAL FLAP	10.00	14.50	8600 (E)	0.625 (.307)	1.188 (1.108)(E)	0.801(E)	3.07	11.08	4.7	2.21	10.39
THRUST (T) VECTOR VANE	5.20	26.00	75,314 (E)(R)	1.875 (2.761) 1.625 (2.074)	3.938 (12.180) 3.875 (11.793)	11.793(E) 9.719(R) 10.106(E) 9.419(R)	14.36	113.87	57.0	"	125.97
NOSE (T) GEAR	5.00	16.75	14,900 (E)	1.250 (1.227) 1.000 (0.785)	1.875 (2.761) 1.594 (1.995)	1.995(E) 1.210(R) 1.976(E) 1.489(R)	6.14	19.85	16.1	"	35.58
MAIN GEAR	6.00	13.13	24,000 (R)	1.250 (1.227)	2.375 (4.430)	3.203(R)	7.36	26.58	8.9	"	19.67
RIGHT RAM AIR SCOOP	2.52	5.33	4,100 (R)	0.437 (.150)	0.938 (.691)	0.541(R)	0.38	1.74	1.2	"	2.65
LEFT RAM AIR SCOOP	1.00	2.88	1,200 (E) (R)	0.375 (.110)	0.625 (.306)	0.196(R)	0.11	0.31	0.35	"	0.77
EXH. DOOR EQUIP. BAY	2.66	5.06	1,300 (R)	0.375 (.110)	0.625 (.306)	0.196(R)	0.29	0.81	0.72	2.21	1.59
CANARD	SEE FIGURE										

(R) = RETRACT
(E) = EXTEND
(T) = TANDEM UNBALANCED ACTUATOR

① BASED ON SWEPT VOLUME (REFERENCE 27)
② CORRECTION FACTOR = 8000/3000 × 83% = 2.21

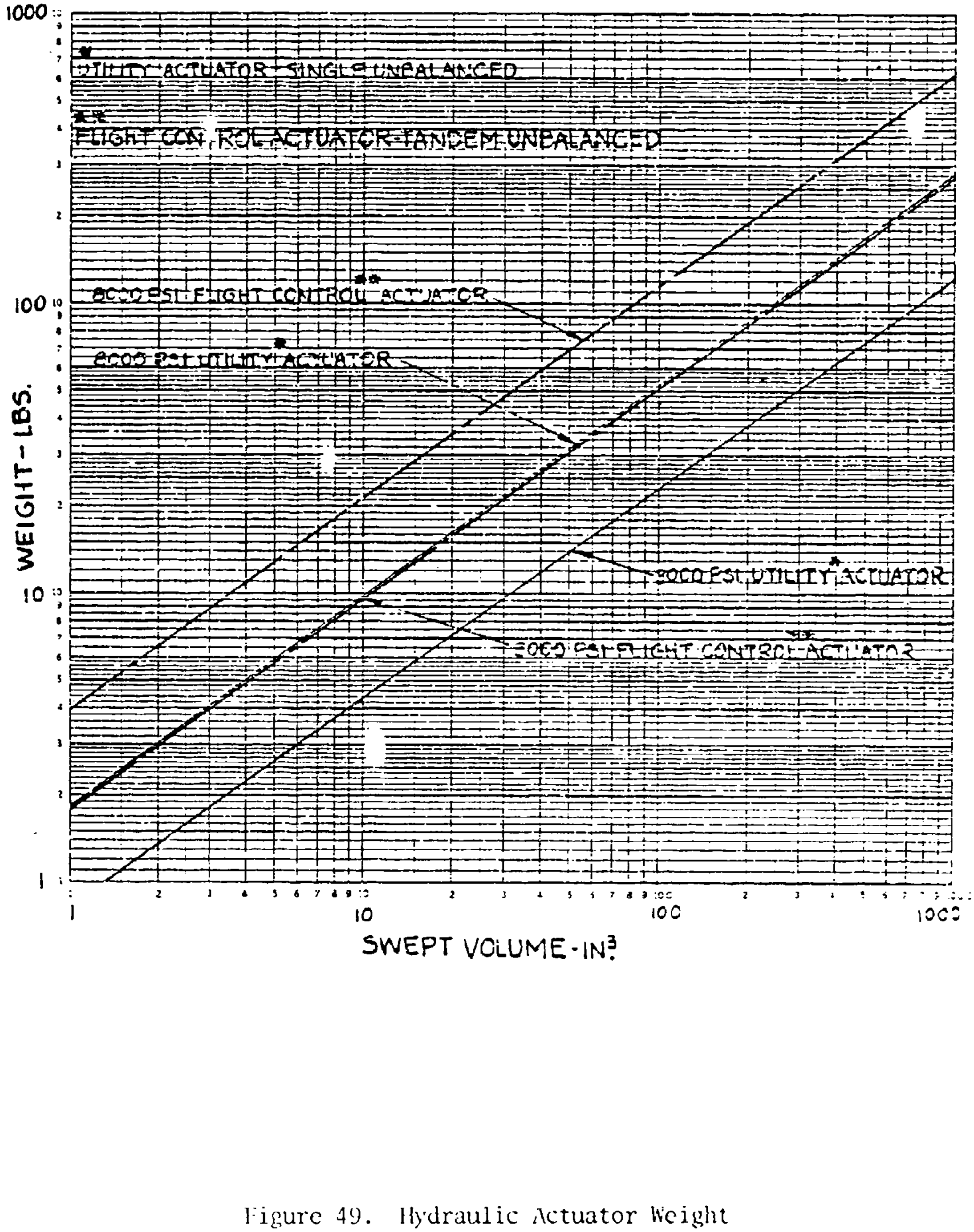


Figure 49. Hydraulic Actuator Weight

volume and/or make up fluid to offset the thermal expansion and contraction of the accumulator's "charged" volume of fluid. If the accumulator involved was relatively large, this could have a significant and serious negative impact on overall system size. For this reason the elements leading to accurate sizing of the accumulator were looked at in considerable detail.

The total energy necessary to start the aircraft II APU/EPU was judged to be about the same as that necessary to start the APU for the F-18 aircraft. Therefore the F-18 aircraft was used as a guide. Tabulated below is the basic data relative to the F-18 APU start motor and its accumulator. The tabulation also includes the comparable known data for aircraft II as well as the items to be determined (TBD) in subsequent calculations:

		F-18	Aicraft II
Fluid		MIL-H-83282	MIL-H-83282
System Pressure	(P)	3000 psi	8000 psi
Accum. Precharge Press/Temp	$(P_{GN})/(T_N)$	1950 psig/75°F	(TBD)/75°F
Accum. Oil Volume Max	(VM)	143 IN^3	(TBD)
Accum. Oil Volume	(VN)	(TBD)	(TBD)
Nitrogen Gas Volume	(V_{GN})	290 IN^3	(TBD)
Motor Displacement	(Δ)	0.364 IN^3	(TBD)
Speed	(S)	14,000 RPM	14,000 RPM
Flow	(Q)	22.06 GPM	(TBD)
Torque	(T)	174 IN-LB	174 IN-LB
Power	(HP)	(TBD)	(TBD)

Aircraft II APU Motor Displacement

$$\frac{P_1}{P_2} \times \Delta_1 = \Delta_2$$

$$\Delta_2 = \frac{3014.7 \text{ PSIA}}{8014.7 \text{ PSIA}} \times 0.365 \text{ IN}^3/\text{REV}$$

$$\Delta_2 = 0.137 \text{ IN}^3/\text{REV}$$

WHERE: P_1 = Rated absolute pressure (PSIA) F-18 motor

P_2 = Rated absolute pressure (PSIA) Aircraft II motor

Δ_1 = Displacement IN^3/REV F-18 motor

Δ_2 = Displacement IN^3/REV Aircraft II motor

Aircraft II APU Motor Flow

$$\frac{\Delta_2 \times S_2}{231} = Q_2$$

$$\frac{0.137 \text{ IN}^3/\text{REV} \quad 14{,}000 \text{ RPM}}{231 \text{ IN}^3/\text{GAL} \times 0.85} = Q_2$$

$$Q_2 = 9.77 \text{ GPM}$$

WHERE: S_2 = Speed (RPM) Aircraft II motor

Q_2 = Flow GPM Aircraft II motor

2 = Motor Efficiency = 0.85

Aircraft II APU Motor Power

$$\frac{Q_2 \times P_2}{1714} = HP_2$$

$$HP_2 = \frac{9.77 \text{ GPM} \times 8014.7 \text{ PSIA}}{1714}$$

$$HP_2 = 45.68 \ h_p$$

$$HP_1 = \frac{22.06 \text{ GPM} \times 3014.7 \text{ PSIA}}{1714 \times 0.85}$$

$$HP_1 = 45.64 \ h_p$$

WHERE: HP_2 = Max. instantaneous power (h_p) Aircraft II motor

HP_1 = Max. instantaneous power (h_p) F-18 motor

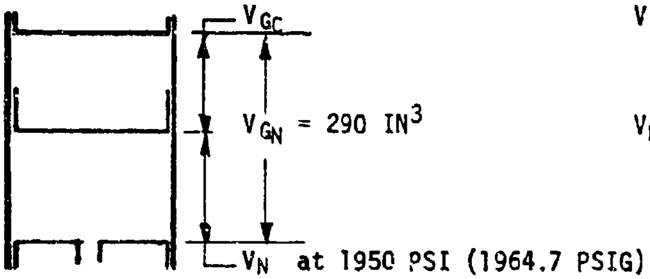
Both accumulators are required to deliver the same energy to their respective APU start motors

Consider the F-18 accumulator:

WHERE: V_{GC} = Charged accumulator gas volume (IN^3)

V_{GN} = Precharged accumulator gas volume (IN^3)

V_N = Charged accumulator fluid volume (IN^3) nominal $= V_{GN} - V_{GC}$



V_{GC} is obtained from:

$$P_{GN}\ V_{GN} = P_{GC}\ V_{GC}$$

WHERE: P_{GN} = Precharged accumulator pressure (PSIA)

P_{GC} = Charged accumulator pressure (PSIA)

$$V_{GC} = \frac{1964.7 \text{ PSIA X } 290 \text{ IN}^3}{3014.7 \text{ PSIA}}$$

$$= 188.99 \text{ IN}^3 \text{ use } 189 \text{ IN}^3$$

$$V_N = V_{GN} - V_{GC} = 290 \text{ IN}^3 - 189 \text{ IN}^3$$

$$= 101 \text{ IN}^3 \text{ (at } 75°\text{F)}$$

To determine the energy available from the F-18 accumulator consider the gas expansion (V_G) as an adiabatic process because the accumulator will be depleted within 1.75 seconds during a normal APU start. The adiabatic relationship is:

$$P_{GC}\ V_{GC}^n = P_{GN_e}\ V_{GN}^n$$

WHERE: $n = 1.4$

$$P_{GN_e} = \frac{(3014.7)\ (189)^{1.4}}{(290)^{1.4}}$$

P_{GN_e} = Pressure after adiabatic discharge to V_{GN} volume

$$= 1655.52 \text{ PSIA}$$

Therefore from the relationship:

$$\frac{T_{GN_e}}{T_{GC}} = \left(\frac{V_{GC}}{V_{GN}}\right)^{n-1}$$

WHERE: T_{GC} = Absolute gas temperature of charged accumulator (°R)

$$T_{GN_e} = (460 + 75) \left(\frac{189}{290}\right)^{0.4}$$

$$= 450.8\ ^oR \quad (-9.2°F)$$

T_{GN_e} = Absolute gas temperature of adiabatically discharged accumulator (°R)

Both the aircraft II and F-18 accumulators are required to deliver the same amount of useable energy. The work of expansion is described by the formula:

$$\text{WORK} = \int_1^2 PdV = \frac{P_2 V_2 - P_1 V_1}{1.0 - n} = \frac{P_{GN_e} V_{GN} - P_{GC} V_{GC}}{1.0 - n}$$

For the adiabatic process,(see above) at or below 3000 psi,n approximates 1.4. Therefore work $= \dfrac{(1655.43)\ (290) - (3014.7)\ (189)}{1.0 - 1.4}$

$$= \frac{480074.7 - 569778.3}{-0.4}$$

$$= 224{,}259 \text{ in-lbs}$$

To deliver this amount of energy at pressures high enough to be useable by the 8000 psi hydraulic motor it is necessary that the ratio of charge pressure (P_{GC}) (8014.7 PSIA) to pressure after adiabatic expansion (P_{GNe}) be equal to, or higher than,that of the 3000 PSI F-18 accumulator. Therefore:
$P_{GNe} = \dfrac{1655.43}{3014.7} \times 8014.7 = 4401.03$ PSIA for aircraft II. To accurately compare the 3000 PSI F-18 accumulator and the aircraft II accumulator, compressibility must be taken into account. Figure 50 plots typical compressibility characteristics for air in the pressure-temperature regime which will be experienced by the accumulator. From the figure it can be seen that the following relationship exists:

$$\frac{P_1 V_1}{Z_1 T_1} = WR = \frac{P_2 V_2}{Z_2 T_2} \quad \text{Where} \quad R = 639.6 \ \frac{\text{IN-LB}_f}{\text{LB}_m\ ^oR}$$

Therefore if the weight (W) of gas in the 3000 PSI accumulator is:

$$\underset{(3000\ \text{PSI})}{W} = \frac{P_1 V_1}{R\ Z_1\ T_1} = \frac{3014.7 \times 189}{639.6 \times 1.049 \times 535} = 1.587 \text{ LB}$$

the charged volume (V_2) is:

$$V_2 = \frac{W\ R\ Z_2\ T_2}{P_2} = \frac{1.582 \times 639.6 \times 1.381 \times 535}{8014.7}$$

$$= 93.59 \text{ IN}^3$$

V_2 presumeably has excessive stored energy by the ratio of the pressures 8014.7 PSIA vs 3014.7 PSIA. Therefore:

$$V_{GC} = V_2 \text{ ratioed} = 93.59 \times \frac{3014.7}{8014.7} = 35.20 \text{ IN}^3$$

$$\underset{(8000\ \text{PSI})}{W} = 1.587 \text{ LB} \times \frac{3014.7}{8014.7} = .5975 \text{ LB}$$

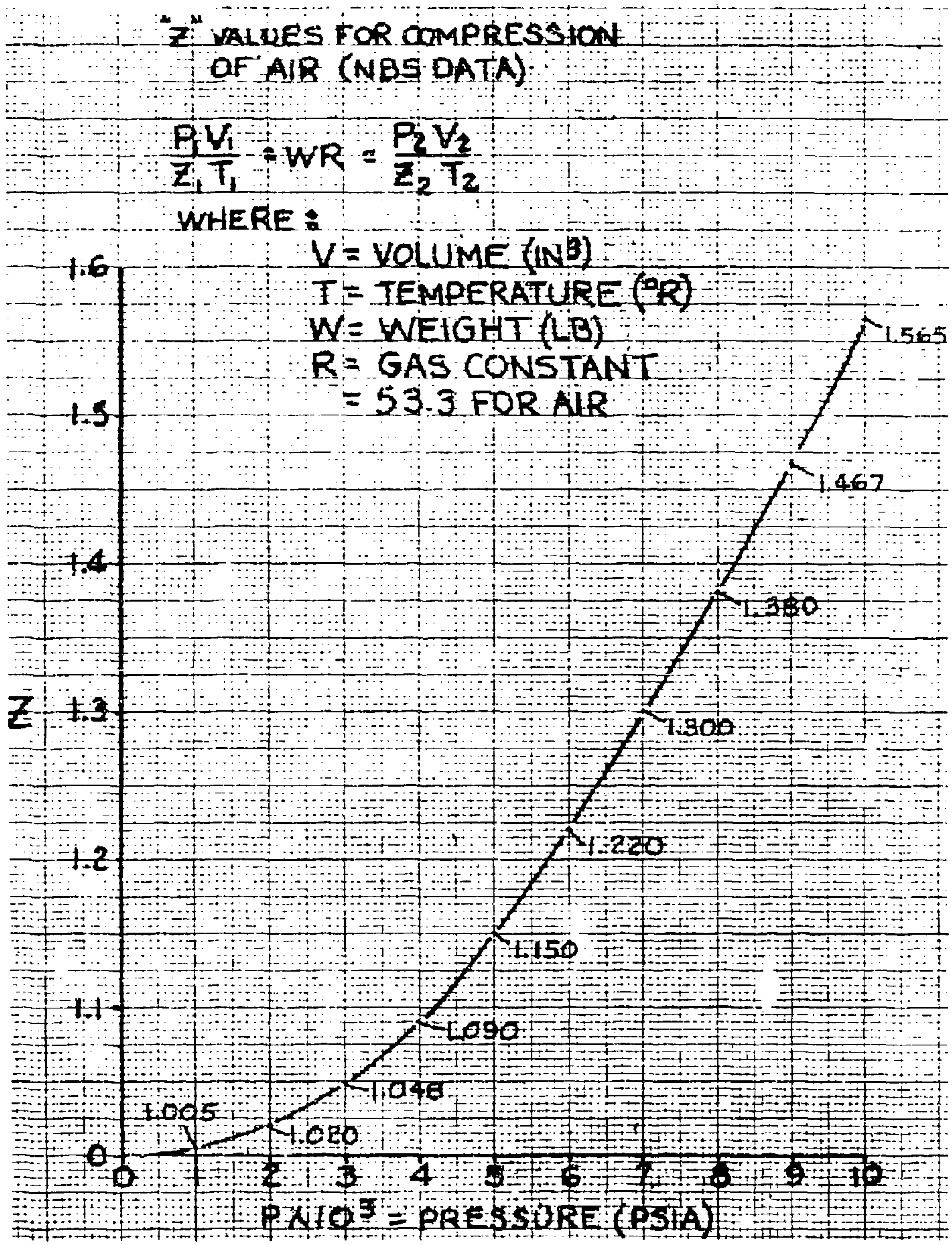


Figure 50. Compression Factor (Z)

Using the relationship $P_{GC} V_{GC}^n = P_{GNe} V_{GN}^n$ and rearranging:

$$V_{GN} = \sqrt[n]{\frac{P_{GC}V_{GC}^n}{P_{GNe}}} = \sqrt[1.4]{\frac{8014.7 \times (35.20)^{1.4}}{4401.03}}$$

$$= 54.02 \text{ IN}^3$$

$$\text{WORK} = \frac{P_{GNe}V_{GN} - P_{GC}V_{GC}}{-0.4} = \frac{4401.03\ (54.02) - 8014.7\ (35.21)}{-0.4}$$

$$= 111049.05 \text{ IN-LB} \quad \text{(too small)}$$

Increase V_{GC} and W by the ratio of required energy to actual energy $\frac{224{,}259}{111{,}049} = 2.02$

Then:

$$V_{GC} = 71.12 \text{ IN}^3$$

$$W = 3.206 \text{ LB}$$

$$V_{GN} = \sqrt[1.4]{\frac{8014.7\ (71.12)^{1.4}}{4401.03}}$$

$$= 109.13 \text{ IN}^3$$

$$\text{WORK} = \frac{4401.03 \times (109.13) - 8014.7\ (71.12)}{-0.4}$$

$= 224{,}303$ IN-LB ($\leq$ the required 224,259 IN-LB)

$V_n = V_{GN} - V_{GC} = 109.13 - 71.12 = 38.01 \text{ IN}^3$

= Fluid expelled

P_{GN} = 8000 PSI accumulator precharge pressure

$$= \frac{P_{GC}\ V_{GC}}{V_{GN}} = \frac{8014.7 \times 71.12}{109.13}$$

= 5223.18 PSIA

The weight of the 8000 PSI precharged APU start accumulator was determined as follows:

WACC = 15 LB (from Figure 51)
WAIR = W = 1.207 LB
Total weight = 16.21 LBS

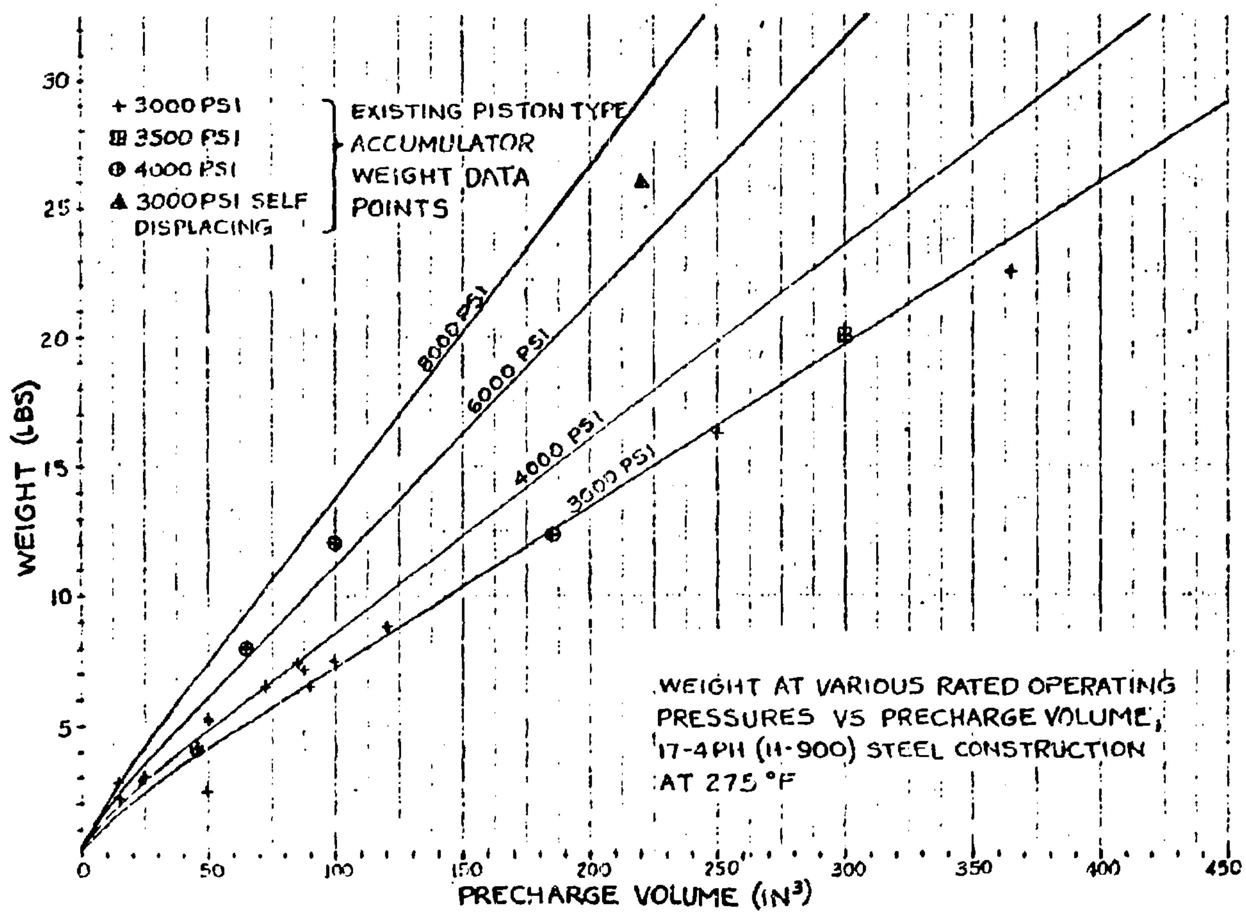


Figure 51. Accumulator Weight

4.2.1.9 APU/EPU Sizing - The APU/EPU performance data used for sizing the APU/EPU, as it was used on aircraft II, was based upon computations made on the following pages and upon the summation of this data as shown in Table 31. Most of the column headings shown on Table 31 are self-explanatory except for the columns headed No. 1 through No. 11. These represent power transmission interfaces between components as shown by the coded ballons on Figure 45. The coded number series is generally arranged in reverse order of power flow from source to final output. Thus column No. 11 in Table 31 is the required output of the unit when encountering the various operating conditions listed under the function column. The first two function listings represent operation as an emergency power unit (EPU) above 20,000 ft altitude and the next four listings represent various conditions of operation as an auxiliary power unit (APU) below 20,000 ft. For a more detailed discussion of several of these operating conditions (functions) see paragraph 4.2.1.1. However, several factors having an impact on APU size, were not exhaustively discussed in paragraph 4.2.1.1 and are expanded here. These factors were:

1. When operating as an EPU the load compressor (LC in Figure 45) was always unloaded therefore, as an EPU, the unit was required to supply only drag load power for the load compressor and either drag load or duty cycle load power for the emergency pump and generator. The reason the freon compressor was unloaded was the fact that, during an emergency, the ram air scoop doors were open to provide the required cooling.

2. When attempting "ground" or "in-flight" engine starts all units drawing power from the AMAD were unloaded (i.e. drag power only) except for the engine PTO.

3. During "ground" engine starts the emergency pump and generator were unloaded, however, during "in-flight" engine starts both units were loaded and remained so until the APU/EPU was shutdown in response to a signal that both primary hydraulic and electrical systems were once again on line and functioning properly.

4. During "ground" functional checkouts both the two AMADs and the AGB could be powered simultaneously using ground air supplied through the ground connection shown on Figure 45. The AMADs were driven by their respective ATS/M and the AGB was driven by partial arc admission of air to the free turbine of the APU/EPU. The ground air system was sized so that, with the engines decoupled, the system could meet the ground checkout simultaneous duty cycled load requirements of all components mounted on both AMADs and the AGB. However, when power for checkout purposes was supplied by the APU the simultaneous load requirements must be reduced. Only one AMAD, with its mounted components, and an AGB, with its emergency generator de-excited, could be driven on a duty cycled checkout load as seen by those components. Using the APU as a power source, all elements of the electrical system could be checked out at 100% of the duty cycled checkout load as could all elements of the hydraulic system except main landing gear actuation functions. The main landing gear could be operated through a complete checkout cycle only if AMAD #2 was being powered by the APU. Also, with the APU as a power source, only one freon compressor could be operated. However, even though this reduced the ECS cooling capacity by 50%, it was adequate to meet all cooling requirements encountered during checkout.

TABLE 31. APU/EPU POWER REQUIREMENTS DEFINITION

FUNCTION	APU/EPU OPERATING MODE	MAXIMUM ALTITUDE (FT)	OPERATING TIME (MAXIMUM SINGLE DURATION)	POWER DELIVERED (HP) AT POINTS CODED IN FIGURE 4.2-4											REMARKS
				1	2	3	4	5	6	7	8	9	10	11 △5	
APU/EPU START [EPU]	GAS GEN.	62,300	< 5 SEC	0	0	0	0	0	0	0	△1 13.48	△1 0.56	△1 8.49	23.96	△1 DRAG POWER COMPUTED ON PAGE 199 △2 BASED ON 15 MIN. EMERGENCY HOUSEKEEPING LOADS ELECT. LOAD ANAL. REF. 8
HIGH ALTITUDE EMERGENCY [EPU]	GAS GEN	62,300	4 MIN.	0	0	0	0	0	0	0	△1 13.48	△2 9.84	△3 70.34	93.66	△3 COMPUTED AS SHOWN ON PAGE 202 △4 BASED ON FULL RATE OF HIGHEST DEMAND SURFACE PLUS 10% FULL RATE ALL OTHERS
GROUND ENGINE START [APU]	GAS TURBINE	5000	40 SEC.	△1 2.61	△1 4.02	△1 16.98	△9 △5 152.74	△3 119.97	△6 179.70	(207) 179.70	△8 △3 239.59	△1 0.56	△1 8.49	(304) 264.51	△5 "INFLIGHT" START POWER = 50% "GROUND" START POWER DUE TO WINDMILL ASSIST △6 ATS/M EFF. = 0.75
GROUND FUNCTIONAL CHECKS [APU]	GAS TURBINE	5000	4 HR	△10 48.20	43.64	△4 76.47	△9 179.05	△11 0	△6 210.65	(242) 210.65	280.86	△1 0.56	△4 48.20	(403) 359.66	△7 BASED ON 5 SEC CRUISE HOUSEKEEPING LOADS WITH ECS LOADS DELETED; FROM ELECT. LOAD ANALYSIS REF. 8
IN FLIGHT ENGINE START [APU]	GAS TURBINE	20,000	60 SEC.	△1 2.61	△1 4.02	△1 16.98	△9 △5 88.95	△5 60.00	△6 104.64	(190) 104.64	△8 139.52	△2 9.84	△3 70.34	(425) 233.73	△8 LOAD COMPRESSOR (LC) EFF. = 0.75 △9 AMAD AND AGB GEARBOX EFF. = 0.94 △10 FROM FIGURE 2-4
EMERGENCY RETURN (APU)	GAS TURBINE	20,000	1.8 HR	△1 2.61	△7 24.87	△4 76.47	△9 110.59	△11 0	△6 130.10	(236) 130.10	△8 173.47	△1 0.56	△4 48.20	(430) 236.41	( ) SEA LEVEL EQUIVALENT POWER PER REF. PAGE 12 △11 ENGINE DECOUPLED FROM AMAD

4.2.1.9.1 (Cont.)

5. In flight engine start requirements were slightly more than 50% of the ground engine start requirements (see column No. 6 on Table 4.2-4). This resulted from the fact that, even though the type of engines used in the study aircraft were incapable of achieving a windmilling engine start in flight, there was a significant windmill assist, during all flight conditions, which could be used to cut the starting power demanded of the APU.

The computations used in the preparation of Table 31 are as follows:

Drag (1) Power at APU/EPU Gear Box Pads

Pump Drag = Rated Power X 0.065

D_{10} = 130.7 HP X 0.065 = 8.49 HP

Gen. Drag = Rated Power X 0.06

D_9 = $\frac{7 \text{ KW} \times 0.06}{.746 \text{ HP/KW}}$ = 0.56 HP

Load Comp. Drag D_8 = Rated Output Power X 0.064

= 210.65 HP X 0.064 = 13.48 HP

Drag (1) Power at AMAD Pads

Pump Drag = $D_{10} = D_3$ = 8.49 HP

Gen. Drag = Rated Power X 0.06

D_2 = $\frac{50 \text{ KW} \times 0.06}{.746 \text{ HP/KW}}$ = 4.02 HP

Freon Comp. Drag = Rated Output Power X 0.064

D_1 = $\frac{30.56 \text{ KW} \times 0.064}{.746 \text{ HP/KW}}$ = 2.61 HP

Air Turbine Start Motor Drag = Rated Output Power X 0.064

D_4 = $\frac{152.98 \times 0.064}{.746 \text{ HP/KW}}$ = 13.12 HP

(1) Power loss with units unloaded or de-excited (see Figure 45)

4.2.1.9.1 APU/EPU Power Requirement for Engine Starting

(Ground - 5000 Ft. Altitude)

P_5 = AMAD power input to engine (HP) (Code 5 Figure 45)

$= 89.5^{②}\ KW/.746\ KW/HP = \underline{119.97\ HP}$

P_4 = Air Turbine Start Motor (ATS/M) input to AMAD (HP) (Code 4 Figure 45)

$$= \frac{2\ D_{10} + D_9 + D_8 + P_5}{\text{AMAD Eff.}} = \frac{16.98 + 4.02 + 2.61 + 119.97}{.94}$$

$= \underline{152.74\ HP}$

$$P_8 = \frac{P_4}{ATS/_M \text{ Eff. x LC Eff.}} = \frac{152.74}{0.85 \times 0.75} = \underline{239.59\ HP}$$

P_{11} = APU/EPU Starting Power Requirement

$$= \frac{P_8 + D_9 + D_{10}}{\text{AGB Eff.}} = \frac{239.59 + 0.56 + 8.49}{0.94}$$

$= \underline{264.51\ HP}$

② From page 3-22 reference 8

4.2.1.9.2 Ground Service Power at AMAD Pads

Pump Output (at pump ports) = Based on full rate on highest demand surface plus 10% of full rate on all others (2 pumps)

P_o = (171-35) 0.1 + 35 = 48.6 KW = 65 HP (3)

Pump Input = P_3 (at AMAD pad) = P_o/Pump Eff. = 65/0.85 = 76.47 HP

Freon Compressor Output (at compressor Ports) = P_{10} = 30.56 KW/.746 HP/KW = 40.97 HP (4)

Freon Compressor Input (at AMAD pad) = 40.97/Comp. Eff. = 40.97/0.85

P_1 = 48.20 HP

Generator Output (at Gen. Terminals) = Based on "warm up and take off-5 sec. load" from load analysis (5)

P_{20} = 27.67 KW/.746 HP/KW = 37.09 HP

Generator Input = P_{20}/0.85 = 37.09/0.85

P_2 = 43.64 HP

AMAD Input $= \dfrac{P_1 + P_2 + P_3}{\text{AMAD Eff.}} = \dfrac{48.20 + 43.64 + 76.47}{0.94}$

$P_4 = \dfrac{168.31}{0.94} = 179.05$ HP

(3) From Table 29

(4) From Figure 10

(5) From Reference 8 with ECS compressor loads deleted

4.2.1.9.3 Emergency Power at APU/EPU Gearbox (AGB) Pads

Pump Power = Required output power/pump eff.

= 44.6 KW/.746 X .85 = 70.34 HP

(1) See Table 29

From Table 31 column No. 11 it can be seen that the EPU power delivery requirement was 93.66 HP and that it was based on the high altitude emergency condition. This same power delivery requirement applied to the partial arc air admission when conducting ground checkouts using ground air supplies at altitudes to 5000 ft. Column 11 also shows that the APU was rated at 430 HP sea level static. This was ratioed from the 236.41 HP it must deliver at 20,000 ft. Based on these ratings the APU/EPU unit weighed 399 lbs. This weight included the hydraulic start motor, permanent magent generator, GN_2 pressurized LOX tank, and the required LOX, the gas generator, and the accessory gearbox shown as "SM", "PMG", "LOX", "GG", and "AGB", respectively on Figure 45.

Other system components such as the load compressor (LC) and the air turbine start motor (ATS/M) were also sized from the data given in Table 31. Column 7 provided the rated output power (sea level static) requirement of the load compressor. This proved to be 242 HP based on the ground functional checkout requirement at a 5000 ft. altitude air base. Column 4 provided the rated output power requirement of the ATS/M. This proved to be 179 HP also based on the ground functional checkout requirement. Based on these load ratings and a 1990 + time frame, the weight of the LC was determined as 75 LB and that of the ATS/M as 32 LB.

4.2.1.10 Plumbing System Sizing - The basic characteristics of the plumbing system are shown in Tables 32 and 33. Most of Table 32 is self-explanatory except for the three columns labelled "Rating at -20°F". These columns represent the flow, ΔP, and velocity ratings, used as a guide for sizing both the trunk and subsystem lines in the aircraft. They were based on -20°F instead of the -40°F temperature, at which the landing gear system must meet full performance, or the +20°F temperature at which all flight control subsystems must also meet full performance. The reason for this was the fact that, designing for -20°F, represented the optimum compromise between the excessively high fluid flow velocities which would result from rating the lines at +20°F, wherein rated flow velocities would approach 90 ft/sec, and the excessively large lines which would result from rating the lines at -40°F. It will be noted from Table 32 that the mean flow velocity in the line never exceeded 50 ft/sec and, in the smaller lines where most of the valves were located, was considerably less.

Table 34 shows the main landing gear system designed at -20°F and assumed the tubing's design rated flow existed in each tube in the system (i.e., branch and trunkline pressure, return and suction tubing). Table 34 also assumed that, since increased pressure represents increased pressure drop, the mean pressure in the system was 4000 psi (i.e., 1/2 of 8000 psi). It can be seen in Table 34 that the total pressure drop in the system, with the line diameters and line lengths indicated, was 2563 psi. This is less than 1/3 of the available system pressure (2666 psi) and was very satisfactory for full performance of the landing gear. However, the branch lines of the system must actually operate at -40°F and will actually flow at some flow less than design rated flow. The actual flow required for the main landing was derived from the "power at operating load-rate" column in Figure 8. The derivation assumed a very conservative effective pressure across the actuator of 4667 psi (see pump sizing paragraph 4.2.1.5) and used the 3.63 HP found in Figure 8. From this the required flow rate per main landing gear retract actuator was found to be 1.33 GPM. This determination used the following formula:

$$GPM = \frac{HP \times 1714}{PSI}$$

Where:

$$HP = 3.63$$
$$PSI = 4667$$

This flow rate was considerably less than design flow (i.e., 5.5 GPM for 3/8 inch diameter return lines and 5.0 GPM for 3/8 inch pressure lines per Table 32). Therefore, even though the pressure drop at a given flow at -40°F increased by roughly a factor of three, the actual pressure drop in the system was considerably less than that shown in Table 34 because of the reduced actual flow. Using an actual flow of 1.33 GPM at -40°F, in the branch lines and design rated flow at +20°F in the trunk lines, the actual pressure

TABLE 32. 3AL-2.5V TITANIUM TUBING DATA

SIZE (IN.) DIA.	SIZE (IN.) WALL	FUNCTION	FLOW AREA (IN^2)	WALL AREA (IN^2)	FLUID VOLUME IN^3/FT. L	FLUID WEIGHT LB/FT. ①	TUBE WEIGHT LB/FT. ②	COMBINED WEIGHT LB/FT.	RATING AT -20°F FLOW GPM	RATING AT -20°F ΔP PSI/FT	RATING AT -20°F VELOCITY FT./SEC.	LINE LENGTH FT./AV	LINE WEIGHT LB/AV	TOTAL FLUID VOL(IN^3)
1/4	.023	RET. & PRESS	.033	.016	.408	.012	.033	.045	1.1	70	10.38	362	16.29	147.7
3/8	.020	RET.	.088	.022	1.056	.033	.044	.077	5.5	48	20.05	114	8.78	120.4
3/8	.035	PRESS	.073	.037	.876	.027	.075	.102	5.0	65	21.97	130	13.26	113.9
1/2	.023	RET.	.162	.034	1.944	.060	.068	.128	17.0	45	33.67	67	8.58	130.2
1/2	.047	PRESS	.129	.067	1.548	.048	.135	.183	12.0	52	29.84	70	12.81	108.4
5/8	.029	RET.	.252	.054	3.024	.093	.109	.202	39.0	43	49.65	44	8.89	133.0
5/8	.059	PRESS	.201	.105	2.412	.075	.212	.287	30.0	51	47.89	58	16.65	139.9
3/4	.027	SUCT.	.380	.061	4.560	.141	.123	.264	28.0	13	23.64	10	2.64	145.6
3/4	.035	RET.	.363	.079	4.356	.135	.159	.294	55.0	29	48.61	—	—	
3/4	.069	PRESS	.294	.148	3.528	.109	.298	.407	45.0	34	49.11	10	4.07	35.3
7/8	.032	SUCT.	.516	.085	6.192	.191	.171	.362	50.0	12	31.09	—	—	
7/8	.042	RET.	.491	.110	5.892	.182	.222	.404	75.0	21	49.01	12	4.85	70.7
7/8	.081	PRESS.	.399	.202	4.788	.148	.407	.555	60.0	25	48.25	—		
1	.036	SUCT	.676	.109	8.112	.251	.220	.471	60.0	9	28.48	10	4.71	81.1
											TOTAL	887	101.53	1126.2

① BASED ON 0.0309 LB/IN^3 FOR FLUID

② BASED ON 0.168 LB/IN^3 FOR TITANIUM

TABLE 33. FITTING DATA

SIZE		WEIGHT (LB)			NO. OF	TOTAL FITTING	TOTAL LINE
DASH NO.	TUBE DIA (IN)	4000 PSI FITTING ①	BURST PRESS. RATIO	8000 PSI FITTING	FITTINGS REQD/AV ②	WEIGHT LB/AV	SUPPORT WEIGHT LB/AV ③
-4	1/4	.033	$\times \frac{24,000}{16,000} =$	.050	145	7.25	1.18
-6	3/8	.043	↕	.065	98	8.72	1.54
-8	1/2	.059	↕	.089	55	9.02	1.52
-10	5/8	.109	↕	.164	41	10.25	1.79
-12	3/4	.167	↕	.250	8	2.72	.47
-14	7/8	.227	↕	.340	5	2.36	.36
-16	1	.395	$\times \frac{24,000}{16,000} =$	.593	4	2.37	.35
				TOTAL →	356	42.69	7.21

① BASED ON TITANIUM ELBOW FITTING AS REPRESENTATIVE OF AVERAGE FITTING WEIGHT I.E., UNIONS AND BULKHEAD UNIONS SLIGHTLY LESS AND TEES AND CROSSES SLIGHTLY MORE.

② BASED ON ONE FITTING FOR EVERY 2.5 FT. OF LINE (REF. TABLE 32)

③ BASED ON 5% OF COMBINED LINE AND FITTING WEIGHT

TABLE 34. LANDING GEAR ΔP AT RATED FLOW AND -20° F

TUBE SIZE DASH NO.	TUBE SIZE DIA. (IN.)	TUBE SIZE WALL (IN.)	FUNCTION	RATED FLOW (GPM)	LINE LENGTH (FT.)	ΔP AT RATED FLOW (PSI/FT.)	TOTAL LINE ΔP (PSI.)	NO. OF FTGS.	FITTING ΔP (PSI/FTG)	TOTAL FITTING ΔP (PSI)	TOTAL SYS. ΔP AT RATED FLOW (PSI)
-4	1/4	.023	RET.	1.1	—	70	—	—	12.3	—	—
-4	1/4	.023	PRESS.	1.1	—	70	—	—	12.3	—	—
-6	3/8	.020	RET.	5.5	14	48	672	6	19.2	115.2	787.2
-6	3/8	.035	PRESS.	5.0	14	65	910	6	16.2	97.2	1007.2
-8	1/2	.023	RET.	17.0	—	45	—	—	34.4	—	—
-8	1/2	.047	PRESS.	12.0	—	52	—	—	23.0	—	—
-10	5/8	.029	RET.	39.0	—	43	—	—	80.0	—	—
-10	5/8	.059	PRESS.	30.0	—	51	—	—	44.0	—	—
-12	3/4	.027	SUCT.	28.0	11	13	143	2	17.4	34.8	177.8
-12	3/4	.035	RET.	55.0	—	29	—	—	40.0	—	—
-12	3/4	.069	PRESS.	45.0	5	34	170	4	33.0	132.0	302.0
-14	7/8	.032	SUCT.	50.0	—	12	—	—	15.0	—	—
-14	7/8	.042	RET.	75.0	6	21	126	3	30.7	92.1	218.1
-14	7/8	.081	PRESS.	60.0	—	25	—	—	26.4	—	—
-16	1	.036	SUCT.	60.0	5	9	45	2	13.0	26.0	71.0
						SYSTEM TOTALS →	2066			497.3	2563.3

drop was found to be 1281.9 psi. From this it would appear that the branch lines could have been reduced to 1/4 inch diameter even though they would have been operating slightly overrated. However, in so doing, the system pressure drop would have jumped by a factor of nearly six and would have become 7345 psi which was unacceptable.

Surge magnitude is a direct function of fluid flow velocity and is associated with fast closing valves. In early aircraft hydraulic system designs, flow velocities were held to 15 ft/sec to control surge induced pressure pulses. In recent years this has been increased to 25 ft/sec. It will be noticed in Table 32 that rated flows in small sized tubes (1/4 in. and 3/8 in. diameters) were held below 25 ft/sec. The larger sized lines, which do not contain fast closing valves and which are little affected by fast closing valve action in the small sized lines branching off of them, were allowed to approach 50 ft/sec fluid flow velocity. In this way the somewhat conflicting requirements of good low temperature operation, high normal temperature transmission efficiency, and low surge pressure generation potential, have been met in an optimum manner for aircraft II.

From Tables 32 and 33 it can be seen that the total weight of the hydraulic plumbing system, complete with fluid, fittings and line supports, is 151.43 lb. Also from Table 32, the fluid volume in the tubing is 1126. in^3 to which was added 89.6 in^3 for the fluid volume contained in the fittings to give a total of 1215.8 in^3.

4.2.1.11 Motor Sizing - The motors were sized using existing 3000 PSI motors as a base point. It was assumed that the output section of the motor (i.e., shoe bearing plate, thrust bearing, output shaft, etc.) would be essentially unchanged, whether the motor was a 3000 PSI unit or an 8000 PSI unit, except [illegible] those changes which would result from being able to operate at slightly ([illegible]) higher speed. It was felt that this would be true because the output power would be the same for either the 3000 or 8000 PSI unit. The input section including porting valve plate, block, block bearing and piston diameters were all reduced in size. The block diameter was not reduced as much as might be thought at first glance, however, because the piston shoe circle, and hence the piston bore circle, would reduce only slightly (i.e. as a function of the slight speed increase). None-the-less, the small block diameter reduction due to the piston diameter decrease allowed for a slightly reduced size for the block bearing and hence justified the slight speed increase already mentioned. Based on these considerations it was projected that any advanced (1990 + time frame) 8000 PSI motor would weigh approximately 85% of its 3000 PSI counterpart (i.e. same power output) and would be rated at a 17% higher speed. Table 35 is a tabularization of data comparing existing 3000 PSI motors to projected 8000 PSI units and Figure 52 plots the 8000 PSI motor data from the table. It can be seen in Figure 52 that, although the larger sized motors showed an almost constant weight/power ratio (i.e. 0.136 lb/output HP). the smaller sized units tended to have a higher ratio. This resulted from

TABLE 35. MOTOR CHARCTERISTICS AND COMPARISONS

3000 PSI MOTOR							8000 PSI MOTOR						
DISPL-ACEMENT IN³/REV	MOTOR WEIGHT ③ (LBS)	SPEED (RPM)	THEO. INPUT FLOW ① (GPM)	ACTUAL INPUT FLOW ① (GPM)	POWER INPUT ② (HP)	POWER OUTPUT ② (HP)	POWER INPUT ② (HP)	ACTUAL INPUT FLOW ① (GPM)	THEO. INPUT FLOW ① (GPM)	SPEED (RPM)	DISPL-ACEMENT IN³/REV	MOTOR WEIGHT (LBS)	CONT-AINED FLUID VOL ⑤ (IN³)
			④			4.60	5.42	1.16	1.08	16,000	0.016	1.20	1.18
0.11	2.00	12,400	5.76	6.19	10.83	9.21	10.83	2.32	2.16	14,095	0.035	1.73	2.59
0.22	3.11	10,000	9.52	10.24	17.92	15.23	17.92	3.84	3.57	11,746	0.070	2.63	5.18
0.44	5.02	8,400	16.00	17.20	30.11	25.59	30.11	6.45	6.00	9,866	0.140	4.09	10.36
0.75	7.26	7,200	23.38	25.14	44.00	37.40	44.00	9.43	8.77	8,457	0.240	5.98	17.76
1.15	9.80	6,400	31.86	34.26	59.96	50.97	59.96	12.85	11.95	7,517	0.367	8.15	27.16
1.50	11.56	5,900	38.31	41.19	72.09	61.28	72.05	15.44	14.36	6,930	0.479	9.80	35.54
2.05	14.45	5,700	50.58	54.39	95.20	80.92	95.20	20.40	18.96	6,695	0.654	12.94	48.40

① VOLUMETRIC EFFECIENCY = 0.93
② OVERALL EFFICIENCY = 0.85
③ HIGH CASE PRESSURE MOTORS (3000 PSI)
④ NO EQUIVALENT 3000 PSI MOTOR FRAME SIZE WHICH IS THIS SMALL
⑤ BASED ON .136 LB/HP EXCEPT FOR THE 3 SMALLEST SIZED MOTORS (SEE FIGURE 52)

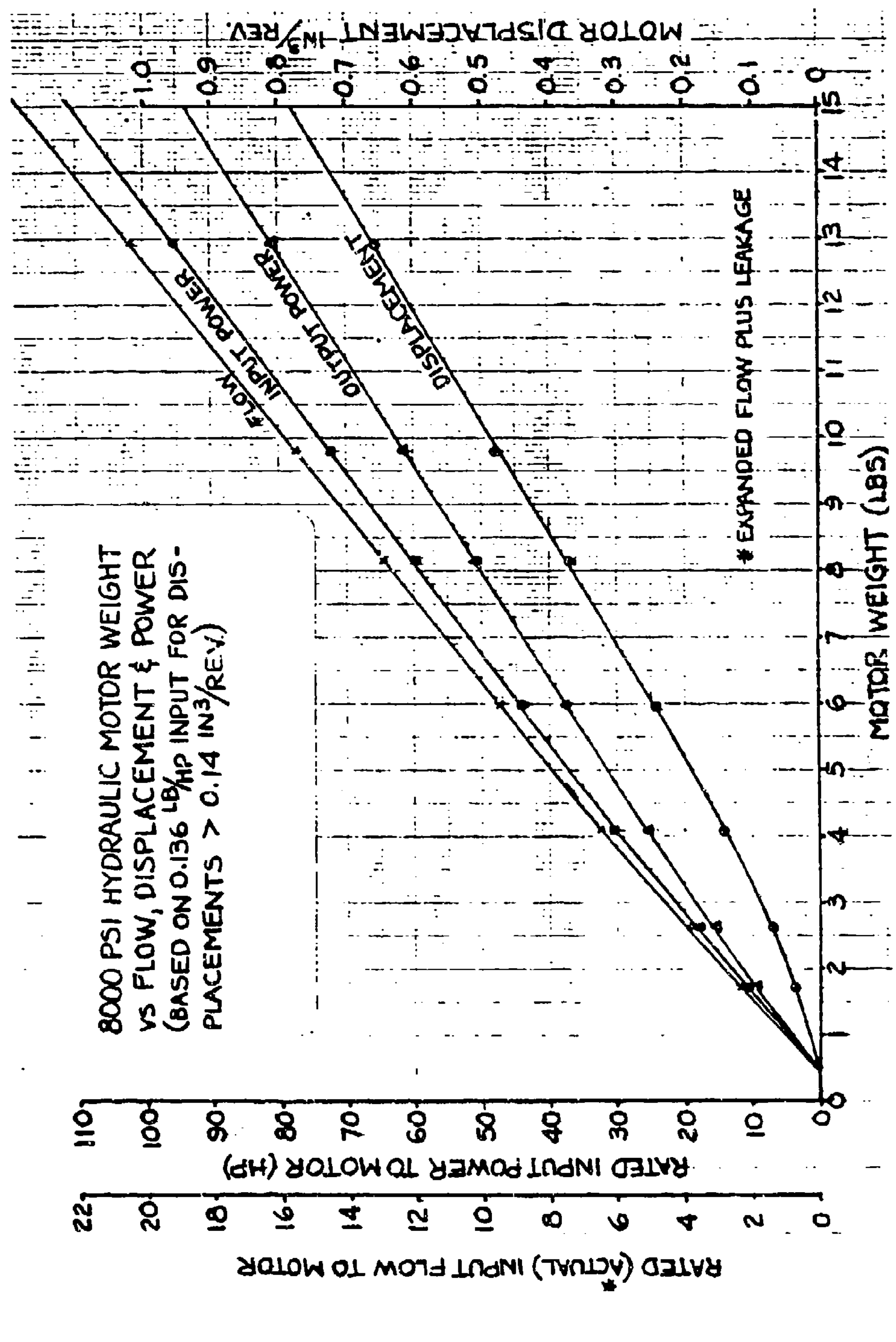


Figure 52. Motor Sizing

the fact that the motor tended to become of "watchwork" size and was impractical to build as small as it could theoretically be. Added to this was the fact that the porting could not reduce proportionately since port sizes for this airplane are not allowed to be less than -4.

4.2.1.12 Reservoir Sizing - The reservoir was sized using MIL-R-8931 paragraph 3.4.1 as a guide. The sizing process followed steps (a) thru (h) shown in paragraph 3.4.1 of the MIL Specification and is listed in approximate order as follows:

(a) An amount of fluid sufficient to ensure that the hydraulic pump inlet pressurization and satisfactory circulation is maintained.

System #1	System #2	System #3
23 IN^3	20 IN^3	10 IN^3

(b) A fluid volume equivalent to 100 percent of the possible net depletion caused by actuator volumetric changes during operation. This data is tabularized in Table 36 and its summation is listed here as follows:

System #1	System #2	System #3
32.72	46.54	0.0

(c) A fluid volume equivalent to 100 percent of the reservoir fluid volumetric change caused by charging all accumulators.

System #1	System #2	System #3
0.0	*38.01	0.0

* See Paragraph 4.2.1.8

(d) A fluid volume equivalent to 130 percent of the volumetric capacity of the largest quantity-measuring type of hydraulic fuse in the system.

No fuses used.

(e) A fluid volume equivalent to the maximum thermal contraction which was expected to occure when the entire fluid content of a recirculating system was exposed to a temperature decrease from 70°F down to -40°F.

		SYSTEM		
		#1	#2	#3
SUMP VOLUME	(a)	23.00 IN^3	20.00 IN^3	10.00 IN^3
ACTUATOR VOL. CHANGES	(b)	32.72 IN^3	46.54 IN^3	________
ACCUMULATOR VOLUME	(c)	________	38.01 IN^3	________

TABLE 36. ACTUATOR SWEPT AND EXCHANGE VOLUMES

ACTUATOR	SYSTEM #1			SYSTEM #2			SYSTEM #3	
	NO. OF ACTUATORS REQ/SYS	VOLUME (IN^3)		NO. OF ACTUATORS REQ/SYS	VOLUME (IN^3)		NO. OF ACTUATORS REQ/SYS	EXCH. AND SWEPT VOLUME
		EXCH.	SWEPT		EXCH.	SWEPT		
CANARD	2 x ½ (T)	7.06	9.89	2 x ½ (T)	7.06	9.89	—	
THRUST VECTOR VANE	2 x ½ (T)	14.36	113.87	2 x ½ (T)	14.36	113.87	—	
EXTERNAL FLAP	2	6.14	22.18	2	6.14	22.18	—	
NOSE GEAR	1 x ½ (T)	3.93	9.93	1 x ½ (T)	3.07	9.93	—	
NOSE GEAR DOWN LOCK	1	.04	.12	1	.04	.12	—	
NOSE GEAR UP LOCK	1	.04	.12	1	.04	.12	—	
NOSE GEAR DOOR	1	.33	1.00	1	.33	1.00	—	
NOSE GEAR DOOR LOCK	1	.04	.12	1	.04	.12	—	0
MAIN GEAR	—	—	—	2	14.72	53.16	—	
MAIN GEAR DOWN LOCK	—	—	—	2	.07	.21	—	
MAIN GEAR UP LOCK	—	—	—	2	.07	.21	—	
MAIN GEAR DOOR	—	—	—	2	.53	1.60	—	
MAIN GEAR DOOR LOCK	—	—	—	2	.07	.21	—	
EXHAUST DOOR	1	.29	.81	—	—	—	—	
RIGHT RAM AIR SCOOP	1	.38	1.74	—	—	—	—	
LEFT RAM AIR SCOOP	1	.11	.31	—	—	—	—	
		32.72	160.09		46.54	212.62		0

(T) = TANDEM ACTUATOR

FUSE VOLUME	(d)	________	________	________
FLUID RECIRCULATING SYSTEM VOLUME CONSISTING OF:	(e)			
1. ACTUATOR SWEPT VOL.		160.09 IN^3	212.62 IN^3	________
2. COMPONENT VOLUMES		320.35 IN^3	304.03 IN^3	127.77 IN^3
3. TUBING VOLUME		489.50 IN^3	470.50 IN^3	166.23 IN^3
4. FITTING VOLUME		40.20 IN^3	36.20 IN^3	13.24 IN^3
TOTAL		1065.86 IN^3	1127.90	317.24

1. See Table 36.

2. See Table 37 which is a master equipment list (MEL) which tabulates the weights and contained volumes of all components making up the Hydraulic System other than actuators, tubing and fittings.

3. See Table 32

4. See Table 33

The differential temperature (ΔT) in varying from 70°F to -40°F is 110°F and the coefficient of thermal expansion (e) for MIL-H-B2382 fluid is 4.6×10^{-4} $IN^3/IN^3/°F$. Therefore the differential volume (ΔV)

$$\Delta V = \Delta T \times C \times \text{SYSTEM VOLUME}$$

For SYSTEM #1 $\quad V = 110 \times .00046 \times 1065.86 = 53.93\ IN^3$

For SYSTEM #2 $\quad \Delta V = 110 \times .00046 \times 1127.90 = 57.07\ IN^3$

For SYSTEM #3 $\quad \Delta v = 110 \times .00046 \times 307.24 = 15.55\ IN^3$

f) A fluid volume equivalent to not less than 5% of the entire system fluid volume, including the reservoir, of a recirculating system in order to minimize the frequency of refilling.

	SYSTEM #1	SYSTEM #2	SYSTEM #3
Volume based on a) through e) above	1119.79 IN^3	1184.97 IN^3	332.79
TIMES 5%	.05	.05	.05
	55.99 IN^3	59.25 IN^3	16.64 IN^3

TABLE 37. MASTER EQUIPMENT LIST (PAGE 1 OF 2 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
1	29 1D01	DISCONNECT-QUICK, PRESS.	-12	3.60	1.50
	1D02	DISCONNECT-QUICK, SUCTION	-16	1.90	3.50
	1F01	FILTER, RETURN	-14	10.20	33.00
	1F02	FILTER, PRESSURE	-12	20.41	35.00
	1F03	FILTER, CASE DRAIN	-6	3.23	6.93
	1H01	HEAT EXCHANGER	-6	3.30	21.20
	1M01	TRANSDUCER, PRESSURE	-4	0.51	0.03
	1P01	PUMP, HYDRAULIC	-10	26.00	6.20
	1P02	PUMP, HYDRAULIC	-10	26.00	6.20
	1R01	RESERVOIR, HYDRAULIC	-14	*	*
	1V01	VALVE, CHECK, SYST #1 RETURN	-14	0.42	0.63
	1V02	, RESTRICTOR CHECK, SUCT. Q.D	-10	0.19	0.40
	1V03	, CHECK, SUCTION Q.D.	-16	0.29	0.75
	1V04	, BYPASS, RETURN FILTER	-14	0.30	1.11
	1V05	, CHECK, RESERVOIR RETURN	-14	0.42	0.63
	1V06	, OVERBOARD RELIEF, RESERV.	-14	0.11	0.85
	1V07	, BLEED, RESERVOIR	-4	0.07	0.04
	1V08	, CHECK, CASE DRAIN	-6	0.05	0.12
	1V09	, CHECK, PUMP PRESSURE	-10	0.22	0.34
	1V10	, CHECK, PUMP PRESSURE	-10	0.22	0.34
	1V11	, CHECK, PRESSURE Q.D.	-12	0.30	0.52
	1V12	VALVE, RELIEF, PRESSURE	-12	1.31	1.20
	1V21	VALVE, CONTROL, N.G. STEERING	-4	0.60	0.80
	1V22	, CONTROL, ARMAMENT	-10	4.80	0.60
	1V23	, SHUTOFF, ARMAMENT	-10	2.12	0.32
1	29 1V24	VALVE, CHECK, ARMAMENT	-10	0.14	0.40
			TOTAL/PAGE	106.91	202.27

TABLE 37. MASTER EQUIPMENT LIST (PAGE 2 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
1	29 1V25	VALVE, SELECTOR, REFUEL RECEPT.	-4	0.50	0.05
	1V26	, SELECTOR, L.H. RAM SCOOP	-4	0.35	0.04
	1V27	, SELECTOR, R.H. RAM SCOOP	-4	0.35	0.04
	1V28	, CHECK, SYS. #1, NOSE GEAR	-4	0.08	0.04
	1V29	VALVE, SELECTOR, NOSE GEAR	-4	0.55	0.06
	1V31	VALVE, 2-WAY RESTRICTOR, N.G. UP LOCK	-4	0.08	0.04
	1V32	, 2-WAY RESTRICTOR, N.G. CYL.	-4	0.08	0.04
	1V33	, 2-WAY RESTRICTOR, N.G. DN LOCK	-4	0.08	0.04
	1V34	VALVE, SELECTOR, N.G. DOOR	-4	0.30	0.05
	1V59	VALVE, SHUTOFF, BRAKE MOTOR	-6	0.71	0.11
	1V60	, CONTROL, R.H. LD'G EDGE FLAP	-6	1.51	0.20
	1V61	, L.H. LD'G EDGE FLAP	-6	1.51	0.20
	1V62	, R.H. INBOARD FLAP	-8	1.82	0.24
	1V63	, L.H. INBOARD FLAP	-8	1.82	0.24
	1V64	, R.H. MIDSPAN FLAP	-6	1.51	0.20
	1V65	, L.H. MIDSPAN FLAP	-6	1.51	0.20
	1V66	, R.H. AILERON	-4	0.65	0.06
	1V67	, L.H. AILERON	-4	0.65	0.06
	1V68	, R.H. LOWER RUDDER	-4	0.55	0.06
	1V69	, L.H. LOWER RUDDER	-4	0.55	0.06
	1V70	, R.H. UPPER RUDDER	-4	0.55	0.06
	1V71	, CONTROL, L.H. UPPER RUDDER	-4	0.55	0.06
	1V72	, SELECTOR, EXHAUST DOOR CYL.	-4	0.35	0.04
	1V73	, CONTROL, L.H. EXTERNAL FLAP	-4	0.93	0.09
	1V74	, R.H. THRUST VECTOR	-4	0.55	0.06
	1V75	, L.H. THRUST VECTOR	-4	0.55	0.06
	1V76	, CONTROL, L.H. PLUG THROAT	-8	2.41	0.27
	1V77	VALVE, SELECTOR, L.H. THRUST REV.	-8	1.82	0.24
	1MH01	MOTOR, HYDRAULIC, N.G. STEERING	-4	1.20	1.18
1	29 1MH02	MOTOR, HYDRAULIC, ARMAMENT	-10	8.15	27.16
			TOTAL/PAGE	32.22	31.25

TABLE 37. MASTER EQUIPMENT LIST (PAGE 5 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN³)
1	29 IMH03	MOTOR, HYDRAULIC, REFUEL RECEPT	-4	1.20	1.18
	IMH08	MOTOR, HYDRAULIC, BRAKE	-6	1.73	2.59
	IMH09	, RH LEADING EDGE FLAP	-4	1.73	2.59
	IMH10	, LH LEADING EDGE FLAP	-4	1.73	2.59
	IMH11	, RH INBOARD FLAP	-8	4.09	10.36
	IMH12	, LH INBOARD FLAP	-8	4.09	10.36
	IMH13	, RH MIDSPAN FLAP	-6	1.73	2.59
	IMH14	, LH MIDSPAN FLAP	-6	1.73	2.59
	IMH15	, RH AILERON	-4	1.20	1.18
	IMH16	, LH AILERON	-4	1.20	1.18
	IMH17	, RH LOWER RUDDER	-4	1.20	1.18
	IMH18	, LH LOWER RUDDER	-4	1.20	1.18
	IMH19	, RH UPPER RUDDER	-4	1.20	1.18
	IMH 20	, RH LOWER RUDDER	-4	1.20	1.18
	IMH25	, LH PLUG THROAT	-8	8.15	27.16
1	29 IMH26	MOTOR, HYDRAULIC, L.H. THRUST REV.	-8	5.98	17.76
			TOTAL/PAGE	39.36	86.85
			* SYSTEM TOTAL	178.49	320.35

* TOTAL LESS RESERVOIR

TABLE 57. MASTER EQUIPMENT LIST (PAGE 4 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
2	29 2D01	DISCONNECT-QUICK, PRESS.	-12	3.60	1.50
	2D02	DISCONNECT-QUICK, SUCTION	-16	1.90	3.50
	2F01	FILTER, RETURN	-14	10.20	33.00
	2F02	FILTER, PRESSURE	-12	20.41	35.00
	2F03	FILTER, CASE DRAIN	-6	3.23	6.93
	2H01	HEAT EXCHANGER	-6	3.30	21.20
	2M01	TRANSDUCER, PRESSURE	-4	0.51	0.03
	2M02	GAGE, N_2 PRESSURE, SYS #3 APU	-4		
	2P01	PUMP, HYDRAULIC	-10	26.00	46.20
	2P02	PUMP, HYDRAULIC	-10	26.00	46.20
	2R01	RESERVOIR, HYDRAULIC FLUID	-14	*	*
	2R02	ACCUMULATOR, SYS #3 APU	-8	*	*
	2V01	VALVE, CHECK, SYS #2 RETURN	-14	0.42	0.63
	2V02	, RESTRICTOR CHECK, SUCT. Q.D.	-10	0.19	0.40
	2V03	, CHECK, SUCTION Q.D.	-16	0.29	0.75
	2V04	, BYPASS, RETURN FILTER	-14	0.30	1.11
	2V05	, CHECK, RESERVOIR RETURN	-14	0.42	0.63
	2V06	, OVERBOARD RELIEF, RESERVOIR	-14	0.11	0.85
	2V07	, BLEED, RESERVOIR	-4	0.07	0.04
	2V08	, CHECK, CASE DRAIN	-6	0.05	0.12
	2V09	, CHECK, PUMP PRESSURE	-10	0.22	0.34
	2V10	, CHECK, PUMP PRESSURE	-10	0.22	0.34
	2V11	, CHECK, PRESSURE Q.D.	-12	0.30	0.52
	2V12	VALVE, RELIEF, PRESSURE	-12	1.31	1.20
	2V24	VALVE, N_2 FILL, APU ACCUMULATOR	-4	0.07	—
2	29 2V25	VALVE, SHUTOFF, APU MOTOR	-4	0.43	0.04
			TOTAL/PAGE	99.55	200.53

TABLE 57. MASTER EQUIPMENT LIST (PAGE 5 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
2	29 2V26	VALVE, DUMP, EPU MOTOR	-4	0.23	0.05
	2V27	, CHECK, EPU MOTOR	-8	0.18	0.19
	2V28	, CHECK, SYS #2 NOSE GEAR	-4	0.08	0.04
	2V29	, SELECTOR, NOSE GEAR	-4	0.55	0.06
	2V31	, 2 WAY RESTRICT. N.G. CYL. UPLOCK	-4	0.08	0.04
	2V32	, 2 WAY RESTRICTOR N.G. CYL.	-4	0.08	0.04
	2V33	, 2 WAY RESTRICT. N.G. CYL DN LOCK	-4	0.08	0.04
	2V34	, SELECTOR, NOSE GEAR DOOR	-4	0.30	0.05
	2V36	, 2 WAY RESTRICT. N.G. DOOR LOCK	-4	0.08	0.04
	2V37	, 2 WAY RESTRICTOR N.G. DOOR CYL.	-4	0.08	0.04
	2V38	, CONTROL, R.H. CANARD CYL.	-4	0.65	0.06
	2V39	, CONTROL L.H. CANARD CYL.	-4	0.65	0.06
	2V40	, CHECK, SYSTEM #2 MAIN GEAR	-6	0.09	0.09
	2V41	, SELECTOR, SYST. #1 MAIN GEAR	-6	0.79	0.08
	2V42	, EMERGENCY DUMP, MAIN GEAR	-4	0.23	0.05
	2V43	, 2 WAY RESTRICT, R.H. MAIN GR. CYL	-6	0.09	0.09
	2V44	, 2 WAY RESTRICT, R.H. MAIN GR. CYL	-6	0.09	0.09
	2V45	, 2 WAY RESTRICT., R.H. M.G. UPLOCK	-4	0.08	0.04
	2V46	, 2 WAY RESTRICT., R.H. M.G. DNLOCK	-4	0.08	0.04
	2V47	, 2 WAY RESTRICT., L.H. M.G. DNLOCK	-4	0.08	0.04
	2V48	, 2 WAY RESTRICT., L.H. M.G. UPLOCK	-4	0.08	0.04
	2V49	, 2 WAY RESTRICT., L.H. MAIN GR. CYL	-6	0.09	0.09
	2V50	, 2 WAY RESTRICT., L.H. MAIN GR. CYL	-6	0.09	0.09
	2V51	, SELECTOR, MAIN GEAR DOOR	-6	0.79	0.08
	2V52	, EMERGENCY DUMP, M.G. DOOR	-4	0.23	0.05
	2V53	, 2 WAY RESTRICT. R.H. M.G. DOOR	-4	0.08	0.04
	2V54	, 2 WAY RESTRICT., R.H. M.G. DOOR	-4	0.08	0.04
	2V55	, 2 WAY RESTRICT., R.H. M.G. DOOR LOCK	-4	0.08	0.04
	2V56	, 2 WAY RESTRICT., L.H. M.G. DOOR LOCK	-4	0.08	0.04
	2V57	, 2 WAY RESTRICT., L.H. M.G. DOOR	-4	0.08	0.04
2	29 2V58	VALVE, 2 WAY RESTRICT., L.H. M.G. DOOR	-4	0.08	0.04
			TOTAL/PAGE	6.33	1.82

TABLE 37. MASTER EQUIPMENT LIST (PAGE 6 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
2	292V59	VALVE, SHUTOFF, BRAKE MOTOR	-6	0.71	0.11
	2V60	VALVE, CONTROL, RH LEADING EDGE FLAP	-6	1.51	0.20
	2V61	, LH LEADING EDGE FLAP	-6	1.51	0.20
	2V62	, RH INBOARD FLAP	-8	1.82	0.24
	2V63	, LH INBOARD FLAP	-8	1.82	0.24
	2V64	, RH MIDSPAN FLAP	-6	1.51	0.20
	2V65	, LH MIDSPAN FLAP	-6	1.51	0.20
	2V66	, RH AILERON	-4	0.65	0.06
	2V67	, LH AILERON	-4	0.65	0.06
	2V68	, RH LOWER RUDDER	-4	0.55	0.06
	2V69	, LH LOWER RUDDER	-4	0.55	0.06
	2V70	, RH UPPER RUDDER	-4	0.55	0.06
	2V71	, LH UPPER RUDDER	-4	0.55	0.06
	2V73	, RH EXTERNAL FLAPS	-4	0.93	0.09
	2V74	, RH THRUST VECTOR	-4	0.55	0.06
	2V75	, LH THRUST VECTOR	-4	0.55	0.06
	2V76	, RH PLUG THROAT	-8	2.41	0.27
	2V77	VALVE, CONTROL, RH THRUST REVERSER	-8	1.82	0.24
	2MH05	MOTOR, HYD., SYSTEM #3 APU		4.09	10.36
	2MH08	, BRAKE	-6	1.73	2.59
	2MH09	, RH LEADING EDGE FLAP	-6	1.73	2.59
	2MH10	, LH LEADING EDGE FLAP	-6	1.73	2.59
	2MH11	, RH INBOARD FLAP	-8	4.09	10.36
	2MH12	, LH INBOARD FLAP	-8	4.09	10.36
	2MH13	, RH MIDSPAN FLAP	-6	1.73	2.59
	2MH14	, LH MIDSPAN FLAP	-6	1.73	2.59
	2MH15	, RH AILERON	-4	1.20	1.18
	2MH16	, LH AILERON	-4	1.20	1.18
	2MH17	, RH LOWER RUDDER	-4	1.20	1.18
2	292MH18	MOTOR, HYD., LH LOWER RUDDER	-4	1.20	1.18
			TOTAL/PAGE	45.87	51.22

TABLE 37. MASTER EQUIPMENT LIST (PAGE 7 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
2	292MH19	MOTOR HYD, RH UPPER RUDDER	-4	1.20	1.18
↕	2MH20	↕ ↕ , LH UPPER RUDDER	-4	1.20	1.18
↕	2MH25	↕ ↕ , RH PLUG THROAT	-8	8.15	27.16
2	292MH26	MOTOR HYD, RH THRUST REVERSER	-8	5.98	17.76
			TOTAL/PAGE	16.53	47.28
			* SYSTEM TOTAL	168.28	304.03

* TOTAL LESS RESERVOIR AND ACCUMULATOR

TABLE 37. MASTER EQUIPMENT LIST (PAGE 8 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
3	293D01	DISCONNECT- QUICK, PRESSURE	-10	2.61	1.04
	3D02	DISCONNECT- QUICK, SUCTION	-12	1.25	2.09
	3F01	FILTER, RETURN	-10	6.54	23.10
	3F02	FILTER, PRESSURE	-10	15.96	19.60
	3M01	TRANSDUCER, PRESSURE	-4	0.51	0.03
	3P01	PUMP, HYDRAULIC	-10	26.00	46.20
	3R01	RESERVOIR, HYDRAULIC FLUID	-10	*	*
	3V01	VALVE, CHECK, RETURN	-10	0.30	0.40
	3V02	,RESTRICTOR, CHECK, SUCTION Q.D.	-12	0.23	0.35
	3V03	,BYPASS, RETURN FILTER	-10	0.25	0.80
	3V04	,OVERBOARD RELIEF, RESERVOIR	-10	0.90	0.70
	3V05	,BLEED, RESERVOIR	-4	0.07	0.04
	3V06	,CHECK, CASE DRAIN	-6	0.05	0.04
	3V07	,CHECK, PUMP PRESSURE	-10	0.22	0.34
	3V08	,CHECK, PRESSURE Q.D.	-10	0.22	0.34
	3V09	,RELIEF, PRESSURE	-10	1.07	0.96
	3V62	,CONTROL, RH INBOARD FLAP	-8	1.82	0.24
	3V63	, ,LH INBOARD FLAP	-8	1.82	0.24
	3V64	, ,RH MIDSPAN FLAP	-6	1.51	0.20
	3V65	, ,LH MIDSPAN FLAP	-6	1.51	0.20
	3V68	, ,RH LOWER RUDDER	-4	0.55	0.06
	3V69	, ,LH LOWER RUDDER	-4	0.55	0.06
	3V70	, ,RH UPPER RUDDER	-4	0.55	0.06
3	293V71	VALVE, CONTROL, LH UPPER RUDDER	-4	0.55	0.06
			TOTAL/PAGE	65.04	97.15

TABLE 37. MASTER EQUIPMENT LIST (PAGE 9 OF 9 PAGES)

HYD. SYST. NO.	MASTER EQUIPMENT LIST NUMBER	COMPONENT TITLE	PORT DASH NO.	DRY WT. (LBS)	FLUID VOL. (IN^3)
3	293MH11	MOTOR, HYD. RH INBOARD FLAP	-8	4.09	10.36
	3MH12	, HYD. LH. INBOARD FLAP	-8	4.09	10.36
	3MH13	, HYD. RH MIDSPAN FLAP	-6	1.73	2.59
	3MH14	, HYD. LH MIDSPAN FLAP	-6	1.73	2.59
	3MH17	, HYD. RH LOWER RUDDER	-4	1.20	1.18
	3MH18	, HYD. LH LOWER RUDDER	-4	1.20	1.18
	3MH19	, HYD. RH UPPER RUDDER	-4	1.20	1.18
3	293MH20	MOTOR, HYD. LH UPPER RUDDER	-4	1.20	1.18
			TOTAL/PAGE	16.44	30.62
			* SYSTEM TOTAL	81.48	127.77

* TOTAL LESS RESERVOIR

(g) A fluid volume equivalent to that resulting from the effects of fluid compression, line and actuator expansion, and external seal deflection. This computation assumed that subjecting all of the fluid in actuators, components, tubing and fittings to 8000 PSI (instead of a mean pressure of 6000 PSI on the pressure side and a mean pressure of 2000 PSI on the return side which would be more realistic) and ignoring the structural expansion of tubing, fittings, components, actuators and seals still gave a reasonable but conservative value for this requirement. Therefore, the volumes subject to high pressure were:

System #1	System #2	System #3
1010.14 IN^3	1023.35 IN^3	307.24 IN^3

Using 2.06 x 10.5 PSI as the bulk modulus of MIL-H-83282 at 250°F (see Figure 8 of AIR 1362), the change in volume which must be accomodated in the reservoir became:

$$\Delta Vol = \frac{\text{Pressurized Vol. } (IN^3) \text{ x Pressure (PSI)}}{\text{Bulk Modulus (PSI)}}$$

$$= \frac{1010.14 \ IN^3 \times 8000 \ PSI}{2.06 \times 10^5 \ PSI} \quad \text{For System \#1}$$

$$= \underline{39.23 \ IN^3}$$

$$= \frac{1023.35 \ IN^3 \times 8000 \ PSI}{2.06 \times 10^5 \ PSI} \quad \text{For System \#2}$$

$$= \underline{39.74 \ IN^3}$$

$$= \frac{307.24 \ IN^3 \times 8000 \ PSI}{2.06 \times 10^5 \ PSI} \quad \text{For System \#3}$$

$$= \underline{11.93 \ IN^3}$$

(h) A fluid volume equivalent to system fluid thermal expansion resulting from 70°F to the maximum operating temperature. (250°F bulk fluid temperature) the fluid volume affected for each system was as follows: (see Item f above)

System #1	System #2	System #3
1175.78 IN^3	1244.22 IN^3	349.43 IN^3

The fluid thermal expansion then became:

$$\Delta Vol \ (IN^3) = \text{coeff. of thermal expansion } (IN^3/IN^3/°F) \\ \times \text{ heated vol. } (IN^3) \times \text{ Temp } (°F)$$

$= (.00046)\ (1175.78)\ (180)$ for system #1

$= \underline{97.35\ IN^3}$

$= (.00046)\ (1244.22)\ (180)$ for system #2

$= \underline{103.02\ IN^3}$

$= (.00046)\ (349.43)\ (180)$ for system #3

$= \underline{28.93\ IN^3}$

Based on the above figures the fluid volume capability of each reservoir, without venting fluid overboard then became:

		SYSTEM #1	SYSTEM #2	SYSTEM #3
SUMP VOLUME	(a)	23.00 IN^3	20.00 IN^3	10.00 IN^3
ACTUATOR VOL. CHANGES	(b)	32.72 IN^3	46.54 IN^3	---------
ACCUMULATOR VOLUME	(c)	---------	38.00 IN^3	---------
FUSE VOLUME	(d)	---------	---------	---------
THERMAL CONTRACTION VOL.	(e)	53.93 IN^3	57.07 IN^3	15.55 IN^3
LEAKAGE ALLOWANCE	(f)	55.99 IN^3	59.25 IN^3	16.64 IN^3
COMPRESSION VOLUME	(g)	39.23 IN^3	39.74 IN^3	11.93 IN^3
VOLUME IN RESERVOIR FOR FLUID WEIGHT COMPUTATION PURPOSES		204.87 IN^3	260.61 IN^3	54.12 IN^3
THERMAL EXPANSION VOL.		97.35 IN^3	103.02 IN^3	28.93 IN^3
FLUID VOLUME CAPACITY OF RES.		302.22 IN^3	363.63 IN^3	83.05 IN^3
RESERVOIR WEIGHT (SEE FIGURE 53)		18.25 LB	18.25 LB	6.15 LB

4.2.1.13 Hydraulic System Weight Summary - A summary of the elements making up the hydraulic system's weight for Aircraft II is shown in Table 38. This summarization included all power generation, distribution, and utilization elements between the power take off at the AMAD and the various power output interfaces points which were common to both Aircraft I and Aircraft II. The total hydraulic system weight is shown in Table 38 as 1197.55 lbs. This compared to the 1362 lbs originally predicted based on a parametric weight analysis (see paragraph 4.2 and Table 28). The 164.45 lb weight reduction of this defined system, versus the parametric evaluation, appeared reasonable. The parametric analysis assumed the use of hydraulic linear actuators and the various bell cranks and levers typically associated with such a system. The defined system used power hinges as the final output device and the weight they represented (approx. 428 lb) was not included as part of the hydraulic system weight. Since hydraulic motors of similar power capabilities, are

TABLE 38. HYDRAULIC POWER GENERATION, DISTRIBUTION, AND UTILIZATION SYSTEM (HPGDUS) WEIGHT

ITEM	BASIS	WEIGHT (LB)* ITEM	WEIGHT (LB)* SUB TOT.
COMPONENTS	SUM OF SYSTEM WEIGHTS - TABLE	428.25	
COMPONENT SUPPORTS	10% OF COMPONENT WEIGHT	42.83	
			471.08
TUBING	DERIVED FROM TABLE	66.90	
FITTINGS	TABLE	42.69	
SUPPORTS	" "	7.21	
			116.80
LINEAR ACTUATORS	LB/ACT. NO. REQ. SOURCE		
CANARD	21.25 x 2 FIGURE	42.50	
EXTERNAL FLAP	10.39 x 4 TABLE	41.56	
THRUST VECTOR VANE	125.97 x 2 " "	251.94	
NOSE GEAR	35.58 x 1 " "	35.58	
MAIN GEAR	19.67 x 2 " "	39.34	
R. RAM AIR SCOOP	2.65 x 1 " "	2.65	
L. RAM AIR SCOOP	0.77 x 1 " "	0.77	
EXHAUST DOOR	1.59 x 1 " "	1.59	
BRACKETS & SUPPORTS	10% OF ACTUATOR WEIGHT	41.59	
			457.52
ACCUMULATOR	PARAGRAPH 4.2.1.8 DISCHARGED & PRECHARGED	16.21	
RESERVOIR	PARAGRAPH 4.2.1.11	43.25	
SUPPORTS	5% OF RESERVOIR AND ACCUMULATOR WT.	2.97	
SYSTEM FLUID		89.72	
			152.15
	TOTAL SYSTEM WEIGHT		1197.55

* DRY WEIGHT EXCEPT FOR FLUID ITEM

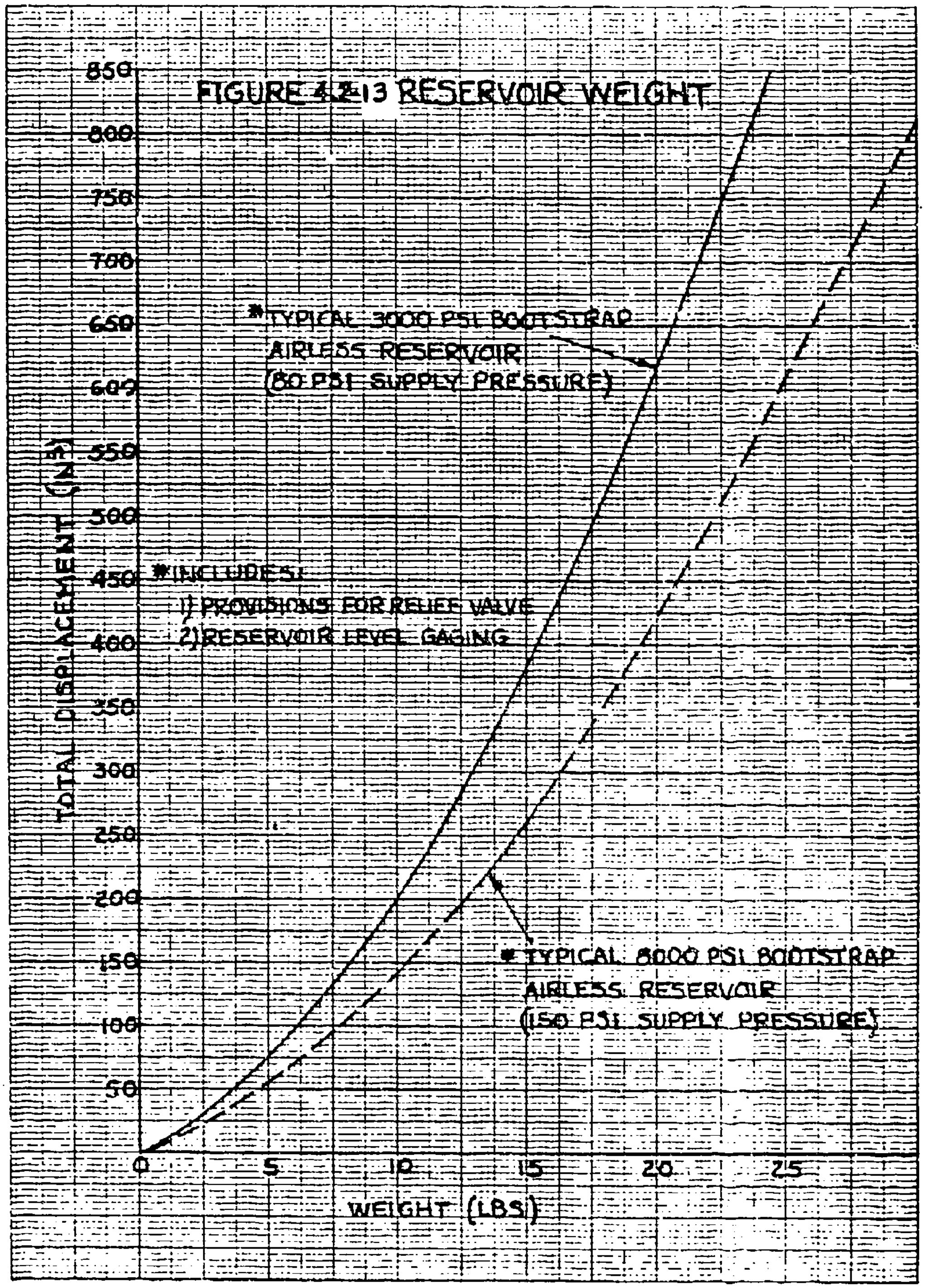


Figure 53. Reservoir Weight

significantly lighter than hydraulic linear actuators, the fact that the weight shown in Table 38 was less than the parametrically projected weight seemed very much in line.

4.2.2 Aircraft II Electrical System General Description - The primary electrical power system for Aircraft II, as shown schematically in figure 54, consisted of two primary AC generators, two transformer-rectifiers, an emergency AC/DC generator and power distribution (bus) system. External electrical power could be applied to the bus system on the ground and a battery provided electrical power to part of the bus system during an engine start without external power.

Two 120/208-volt, 400 Hz generators were the primary source of electrical power. Each generator was powered by separate, engine-driven, remotely-mounted gearboxes. The two generators were connected for split bus, non-synchronized operation. This meant that with both generators operating, each generator supplied power independently to certain aircraft busses. If one generator failed, it dropped off the line; and, at the same time, power from the remaining generator was provided to the busses of the failed (or turned off) generator. Current protection was provided to prevent a fault in one generator system from shutting down both generators; and either generator was capable of supplying power to the entire system. Each generator was activated auto matically when its control switch was in the ON position, and the generator was connected to its busses when voltage and frequency were within prescribed limits (approximately 50% engine rpm). A protection system within the generator control unit protected against damage due to undervoltage, overvoltage, over- and underfrequency, feeder faults, and generator locked rotor. If a fault or malfunction occurred, the generator control unit removed the affected generator from its busses. Except for an underfrequency condition, the control switch of the affected generator must be cycled to bring the generator back on the line after the fault or out-of-tolerance condition cleared. If the generator dropped off the line due to underfrequency and the prescribed frequency was restored, the generator would come back on the line automatically. A generator might be removed from its busses at any time by placing the generator control switch to OFF.

The electrical power generation and distribution system (EPGDS) was designed to provide electrical power to using subsystems of the aircraft during conditions of normal and emergency operation. Subsystems included avionics/instruments, environmental control systems, fuel, hydraulics, landing gear, lighting, propulsion and weapons delivery. The system was specifically designed to the following requirements:

(1) No single failure of the electrical bus will cause loss of the aircraft.

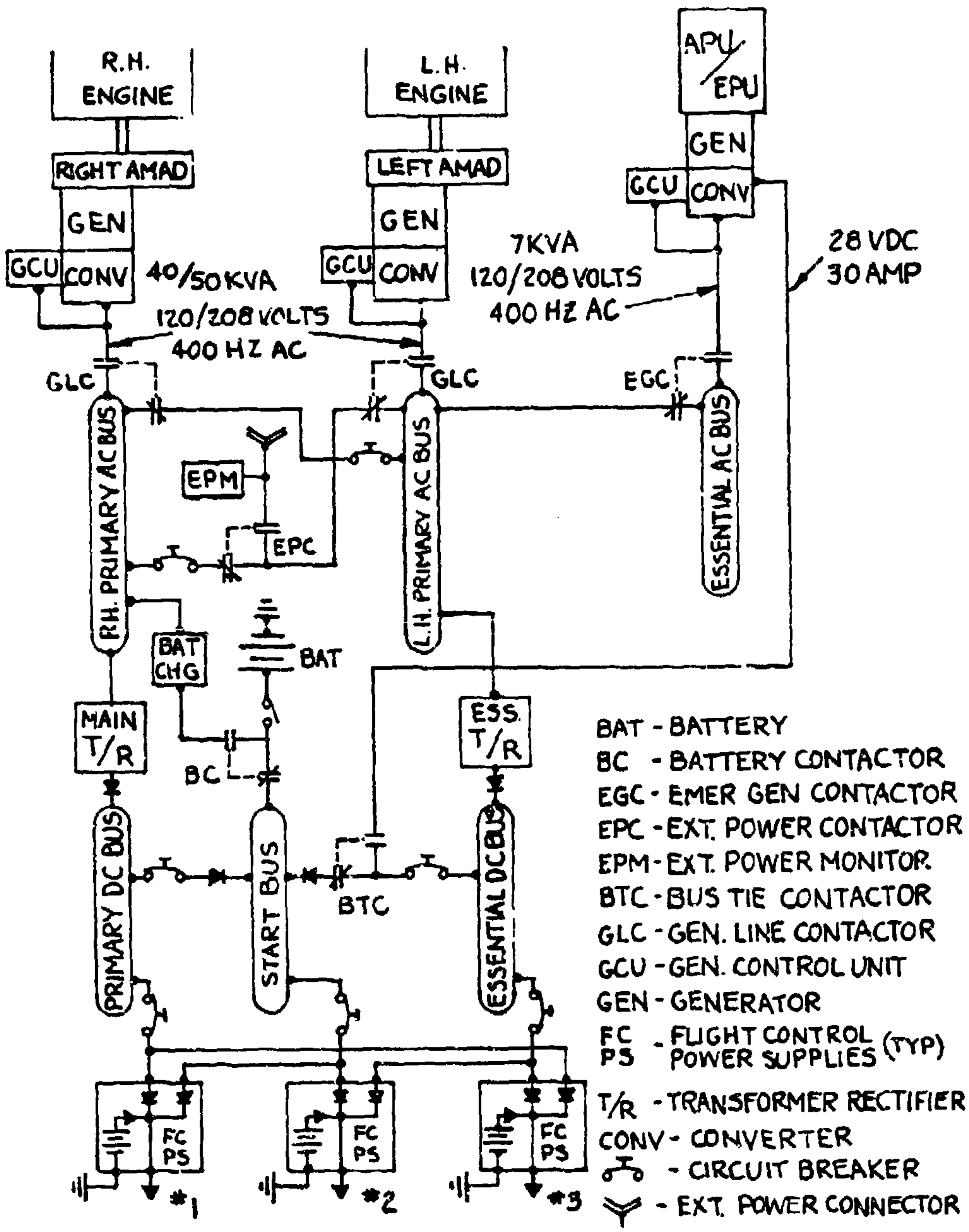


Figure 54. Aircraft II Electrical Power Generation System Schematic

(2) Electrical power characteristics for electrical using equipment are equal to, or better than, MIL-STD-704.

(3) Design and installation of the electrical system conforms to the requirements of specification MIL-E-25499 as specified.

(4) Triplex redundancy provides assurance of an uninterrupted power supply for the three fly-by-wire (FBW) computer channels of the flight control system.

4.2.2.1 DC Electrical Power - Two 25-ampere, upregulated, static transformer-rectifier (TR) units were provided to supply the DC power requirement of approximately 19 amperes. Normally, each TR unit would deliver 50% of the total DC load, at a nominal operating voltage of 27.5 volts to its bus. In the event one transformer-rectifier failed, the other transformer-rectifier would power the entire DC system. The outputs of the TR's were connected in parallel; however, protection was provided through the use of circuit protectors and rectifier elements in the feeders. The circuit protectors eliminated the possibility of a battery bus fault resulting in the loss of both TR units. The rectifier elements protected against a failure in one TR unit affecting the other. No cockpit warning of single transformer-rectifier failure was provided.

With the AC power input between 195 and 210 volts, line-to-line, the two TR units conformed to the requirements of MIL-P-26517. At this input, the unregulated output was within the limits of 25.1 to 29.0 volts. During normal operation, with each unit sharing the total load, the output voltage might vary between 25.6 and 27.5 volts. Internal radio noise filtering of the TR unit met all provisions of MIL-I-26600 and MIL-I-6181D. A variable-speed, constant-volume blower permited normal in-flight cooling by forced air.

4.2.2.2 Emergency Generator - Emergency electrical power was provided by an all altitude APU/EPU driven AC/DC generator that was sized to provide sufficient power to ensure return of the aircraft to its base in the event of loss of primary AC power. The emergency electrical system was separate from the primary electrical system. If either or both main generators were inoperative or both transformer-rectifiers failed, or some combination of faults occurred. the emergency generator was activated and attached to the essential AC/DC busses.

The emergency generator was an air-cooled, brushless, single-bearing machine with a nominal speed range of 12,000 rpm. It was blast-cooled throughout the entire flight profile and over a pressure altitude range from sea level to 60,000 feet. Cooling air was provided by the environmental control system either normally or through the Ram Air Scoops provided (see Figure 10). The system was designed to operate for a minimum of five minutes without cooling

air to ensure availability of electrical power in the event of a temporary loss of the cooling system.

The generator control unit provided the necessary functions for voltage regulation and control of the emergency power system. A static-type voltage regulator was incorporated in the control unit to provide steady-state and transient control of the output voltage at the essential AC and DC busses within the limits of specification MIL-STD-704. Power for buildup and operation of the system was provided by a PMG integral with the generator.

To connect the generator to the essential busses, the control unit provided two 1-ampere, 28-volt DC outputs for closing power transfer relays. Since the emergency power system was a "last-ditch" source of electrical power, the system protection was kept to a minimum. The control unit had an undervoltage sensing function that disconnected the emergency generator from the load bus, after any phase voltage fell below 70 volts, to protect utilizing equipment from damaging exposure to the decaying voltage. Emergency power reset was accomplished by deexciting and subsequently reexciting the generator.

In flight, activation of the APU/EPU was automatic when loss of primary power was sensed and the APU/EPU control switch was in the AUTO position. An ON switch position was provided to activate the APU/EPU, excite the generator, and attach it to the essential buses even when primary power was available. When the emergency generator contactor is picked up, th. main DC bus was isolated from the essential DC bus by deenergizing the bus-tie contactor. The APU/EPU system would provide rated speed and power within approximately three seconds from the time when it has been activated.

Annunciators and a control switch were provided for the emergency generator system. The switch was a standard three-position switch, guarded in the AUTO (normal) position, and had the following control functions:

> "OVERRIDE" - Overrides undervoltage trip protection function. Pilot option not recommended due to potential for damage to power utilizing equipment.
>
> "AUTO" (Normal) - Normal switch position for all ground and flight modes. When APU/EPU is driving generator at normal speed, the generator shall be automatically excited and connected to essential busses. APU/EPU ELEC advisory legend (green) illuminates when emergency generator connected to essential bus. EMERG GEN FAIL warning legend (red) is displayed when the APU/EPU is on and the emergency generator has tripped off line.

"OFF" (Reset) - Provided for emergency generator to be deenergized and idsconnected from the essential busses via the essential bus contactors. Permitted generator reset by momentarily positioning switch to OFF and subsequently repositioning to AUTO. EMERG GEN OFF caution legend (yellow) illuminated.

4.2.2.2 External Power System - The aircraft-mounted external power system consisted of a standard external power receptacle, external power contactor, and power monitor unit to control application of external power to the aircraft. The power monitor unit prevented actuation of the external power contactor if phase rotation, voltage, or frequency of the external power system were not within specified limits. Only three-phase, 115/200-volt, 400 Hz, AC power was required from the ground power source to energize the aircraft bus system. All DC power was supplied by conversion units mounted in the aircraft.

Control of the external electrical power was by means of an external power switch located in the cockpit. The switch had ON and OFF positions. In the ON position and with no generator power, the external power supplied the total aircraft load. With either engine operating, the external power was automatically disconnected and the total load was supplied by the operating generator(s). With the switch in the OFF position, external power could not be supplied to the aircraft busses. The OFF position also provided reset capabilities in the event external power could not be applied to the air vehicle due to improper voltage or frequency tolerance.

4.2.2.3 Battery System - A battery system was provided in the aircraft to supply power to functions required in support of ground-starting the APU/EPU without the need for external electrical power. A secondary purpose was to provide limited emergency capability in the event of loss of all electrical power.

Normally, the DC start bus was supplied 24-volt DC power from the DC essential bus via the transformer-rectifier units when ground or vehicle power was available. A nickel-cadmium, 24-volt battery supplied power to the start bus when essential bus power was not available. Use of 24-volt DC power was dedicated to safety and special start functions, including fire detection and extinguishing, because the use of AC power offers weight advantages. The battery was maintained in a charged state by its own dedicated battery charger and only specific battery-utilizing systems would be exposed to battery-charging voltages. Load requirements of the DC start bus that determined the size of the battery are summarized as follows:

System	DC Start Bus Loads (Watts)
Intercom	30
Communications	86

No. 1 Digital Control Unit Logic	190
Multimode Display Unit No. 1	80
Display Electronic Unit No. 1	240
APU/EPU Control Unit	70
Aural Warning System	10
Cockpit Utility Lights	8
	Total = 714 Watts

The battery selected was a 24-volt, 3 ampere-hour battery that would provide 60 amperes for two minutes at 0°F and for 40 seconds at -20°F. A switch located on the cockpit electrical power panel provided ON-OFF control of the battery power. With the aircraft busses powered by external power on the air vehicle generating system, and if the battery switch was ON, the battery would be charged by the battery charger.

4.2.2.4 Electrical Load Analysis - The housekeeping loads, as shown in Reference 8 provided the basis for determining the AC and DC electrical power required for Aircraft II during various operating modes. These loads provided the design criteria for sizing and selecting the electrical power generator, control and distribution equipment for the aircraft. The maximum demand for primary AC power was on the order of 50 KVA and occurred during the combat portion of the mission. Since the majority of the electrical loads were of the continuously operating type (15 minutes or longer), and the peak 5 sec. load was 74 KVA (70.5 KW x .095 pf). The two primary generator ratings were determined to be 40/50 KVA each.

4.2.2.5 Emergency Generator - The emergency loads of the AC/DC load analysis represented the loads supplied by the APU/EPU-driven generator via the essential busses. The emergency generator was sized at 7 KVA based on the emergency contiuous housekeeping load shown on page A-2 of Appendix E. Of this .75 KVA was assumed to require DC power.

4.2.2.6 Fly-By-Wire Power Supply - To supply power for the fly-by-wire (FBW) flight control system, three flight control power supply (FCPS) units were provided. Each FCPS unit was isolated from the others by diodes and was dedicated to one of the three flight control computer channels. During normal operating modes of the electrical power system, each FCPS unit received 28 volts DC from two of the three DC busses (main, essential, and the battery bus), then passed it on to the three flight control channels. To ensure that there were no voltage transients or interruptions as a result of switching operations normal to aircraft power systems, each FCPS unit

also contained a 24-volt, 1.4 ampere-hour battery which was connected to its input DC power bus. The FCPS batteries were continually charged by the circuitry in the control unit, but were not sized to provide an emergency source of power for a sustained period of time.

4.2.2.7 Primary AC Generators - As determined by the load analysis of Reference 8 (pages A-1 and A-2 with ECS loads deleted. See Paragraph 4.2.2.4), the generating system of Aircraft II was rated at 40/50 KVA of 115/200 volt, three-phase, 400 Hz power. Based largely upon weight, maintainability, and availability, two variable speed constant frequency (VSCF) generators were selected for tradeoff evaluation: (1) the DC-link generating sytem; and (2) the cyclo-converter generating system.

The primary advantage of both systems was that they eliminated the constant speed mechanical/hydraulic drive of the conventional integrated drive generator (IDG) system, and coupled the engine gearbox directly to the VSCF generator. With variations in engine speed, the frequency of the generator output was converted to a constant output frequency of 400 Hz by means of an electronic converter. By replacing the mechanical/hydraulic constant speed drive (CSD) with a sold-state power converter, it was felt that the reliability, maintainability, and life cycle cost of the generating system would be significantly improved (see Reference 14 and 23).

The basic difference between the DC-link approach and the cycloconverter was the type of electronic switch used in the converter and the type of input to the converter. In the case of the cycloconverter, the input was a multiphase, varying frequency waveform. The DC-link system, as the name implies, used a DC voltage as the converter input. The electronic switch in the cycloconverter was an SCR, while transistors were used in DC-link systems as the switch elements. Table 39 compares the different types of typical 30/40 KVA generating systems with respect to weight, efficiency, and operating temperatures.

Table 39

Aircraft Generating System Comparisons

	IDG/CSD	Cycloconverter	DC-Link
Input Oil Temp Limitation	150°F	80°C	120°C
Efficiency (30/40 KVA)	66.4%	71.4%	76.3%
Weight (30/40 KVA)	79 lbs	77 lbs	82 lbs

For both VSCF systems, a high-speed gearbox and a narrow speed range were desirable to minimize overall system weight. As the speed range decreased (i.e. 1.8:1), for a fixed maximum upper speed, the generator weight decreased. Reliable high-speed (27,500 rpm) gearboxes and/or speed increasers were within the state-of-the-art and being flight-tested on the F-18 and F-5G aircraft.

Temperature and type of cooling medium can have a direct impact on the choice of the VSCF system. This criteria directly relates to the temperature capability of the power switching components. The DC-link system power transistors can operate at a higher temperature limitation (120°C) than the thyristors of the cycloconverter system (80°C). Two predominant cooling methods were employed by the system suppliers: (1) spray-oil cooling by the DC-link system; and (2) conduction-oil cooling by the cycloconverter system. Each of these techniques had its advantages and limitations with regard to system weight, cost, efficiency, and reliability. It also had some bearing on the aircraft oil management system.

Both systems produced a quality of electrical power that met or exceeded the requirements of MIL-STD-704. Technological advances in the area of electrical generating systems have been largely directed toward the development of solid rotor generators, using rare earth samarium cobalt magnets, and developing a microprocessor to perform all the control circuit functions with fewer electronic components. Implementation of most of the new hardware advances improved VSCF size, weight, cost and failure rate very little. Only when most of the control circuits are replaced by a microprocessor could significant improvements be realized.

In spite of the lower weight and better part load efficiency of the cyclo-converter approach and the fact that it was chosen for Aircraft I AC load requirements (see Paragraph 4.1.6), the DC Link approach was selected for Aircraft II. This selection was made because of the higher temperature tolerance of the DC Link approach considering the fact that the elaborate evaporative cooling techniques necessary for Aircraft I would not be used in Aircraft II (no inverters) and that fact that the cycloconverter's portion of the total system output was only 25 KVA versus Aircraft II's output of 50 KVA.

4.2.2.8 Fly-By-Wire Control System Arrangement - The fly-by-wire system in Aircraft II was essentially identical to the signal system used in Aircraft I. The Aircraft I arrangement is shown in Figure 37. Like the Aircraft I arrangement, the Aircraft II system employed five microprocessors in the wing and tail, three microprocessors in the nose, and two redundant flight data computers remotely located from each other in the fuselage. Also, like the Aircraft I system, Aircraft II used electro-optical signal interties and the signal transmission lines (see Figure 35) connecting similar components (i.e., motors, actuator clutches, etc.). In the case of Aircraft I, the signal inputs to the actuator were typically fed through an inverter while the comparable item in Aircraft II was a servo valve. In either instance, the signal power and signal characteristics were considered essentially identical. The only significant difference between the signal system used on Aircraft II versus that used on Aircraft I was the fact that Aircraft II had only three power supplies (see Figure 55), whereas Aircraft I had four (see Figure 37).

In spite of this difference the impact on weight and reliability was negligible. Even though Aircraft I had four power supplies, power supply #3 in Aircraft II handled the same power demand as power supplies #2 and #3 in Aircraft I so the power supply weight was considered essentially unchanged. With reference to reliability, Table 40 shows the control and power supply interrelationship on Aircraft II and illustrated that reliability was not impacted. As a typical example, Table 40 illustrates that, if any two power supplies failed to the e wing functions, roll, pitch, and lift control would be maintained. Referring to the table, if power supplies #2 and #3 failed, roll control would be maintained with the right hand outboard flap (OTE-RH) supplemented by the roll function of the left hand and right hand midspan trailing edge flaps (MSTE-LH and MSTE-RH). In a similar manner the pitch and lift functions would be maintained by the right and left hand inboard trailing edge surfaces (ITE-RH and ITE-LH) supplimented by the two midspan trailing edge surfaces. Based on the above analysis, and considering that any differences resulting from the fact that the two systems were supplied different types of power (i.e. 270 VDC for Aircraft I and 115/200 VAC for Aircraft II) would appear as deltas in the distribution system, the fly-by-wire system was eliminated as an item in the trade study.

4.2.2.9 Aircraft II Wiring System - The weight of the Aircraft II wiring, subject to trade, was determined in a manner similar to that used for Aircraft I as discussed in paragraph 4.1.9.2. The wiring weight, using this technique, was found to be 23.0 Lb.

4.2.2.10 APU/EPU Sizing - The Rocketdyne Division of Rockwell had performed extensive development work on all altitude APU/EPU and/or super integrated power units (SIEPU). Therefore, Rocketdyne was asked to evaluate and size an optimum APU/EPU. Four configurations were evaluated, of which, the unit shown in figure 55 was selected. This unit differed from, and/or expanded upon, the unit shown in figure 45 in certain areas. The significant differences between the two were the addition of two heat exchangers and a fuel accumulator to the final selected configuration (figure 55). The weight of the configuration shown in figure 55 was determined as 399.0 Lb and its envelope was 42 in. X 36 in. X 20 in.

4.2.2.11 Aircraft II Electrical System Weight - The weight of the major electrical components making up that portion of the electrical system subject to trade are shown in Table 41. In general these were the power generation and distribution components shown in the schematic of figure 54. Table 41 shows that the total electrical system weight subject to trade was 287 Lb.

4.2.2.12 Aircraft II Total System Weight Subject to Trade - Table 42 shows the total Aircraft II weight subject to trade and lists it as 2,319 Lbs.

TABLE 40. AIRCRAFT II CONTROL AND POWER SUPPLY INTERRELATIONSHIP

	WING ACTUATION FUNCTION (SEE CODE 3)															
	OTE LH	OTE LH	MSTE LH	MSTE LH	MSTE LH	ITE LH	ITE LH	ITE LH	ITE RH	ITE RH	ITE RH	MSTE RH	MSTE RH	MSTE RH	OTE RH	OTE RH
HYDRAULIC POWER SYS. NO.	1	2	1	2	3	1	2	3	3	2	1	3	2	1	2	1
DRIVEN BY AMAD (ENG) NO.	1	2	1	2		1	2			2	1		2	1	2	1
DRIVEN BY APU/EPU					3			3	3			3				
MICRO-PROCESSOR (1)	4	2	1	2	4	1	2	4	4	3	1	4	3	1	3	1
ELECTRICAL (2) POWER SUPPLY	3	2	1	2	3	1	2	3	3	2	1	3	2	1	2	1

(1) SEE FIGURE 4·21 FOR WING "W" MICROPROCESSOR NUMBERING SYSTEM
(2) SEE FIGURE 4.2-13 FOR ELECTRICAL POWER SUPPLY NUMBERS

(3) OTE = OUTBOARD TRAILING EDGE SURFACE
MSTE = MIDSPAN TRAILING EDGE SURFACE
ITE = INBOARD TRAILING EDGE SURFACE (AILERON)
RH = RIGHT HAND
LH = LEFT HAND

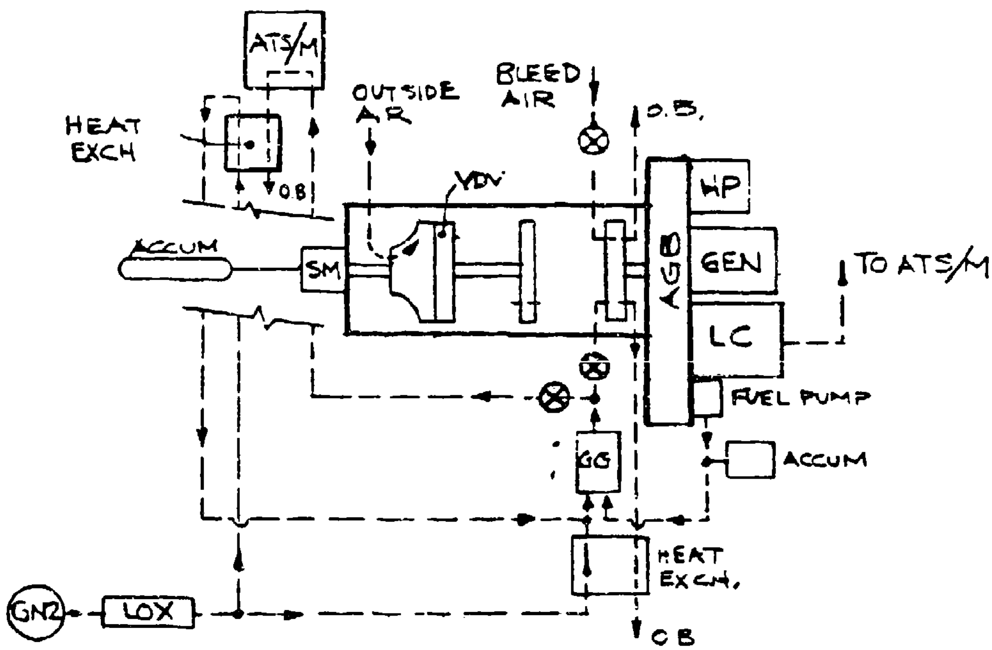


Figure 55. Final APU/EPU Configuration

TABLE 41. AIRCRAFT II ELECTRICAL SYSTEM WEIGHT

EQUIPMENT ITEM ①	QUANTITY PER A/C	UNIT WEIGHT (LB)	TOTAL WEIGHT (LB)
PRIMARY GENERATOR/CONVERTER (40/50 KVA)	2	88.00	176.00
TRANSFORMER RECTIFIER (25 AMP)	2	4.00	8.00
EMERG. GENERATOR/CONVERTER (7 KVA-AC, 30 AMP DC)	1	18.00	18.00
BATTERY (3AMP-HR, 24 VOLT)	1	10.50	10.50
BATTERY CHARGER (10 AMP)	1	4.00	4.00
FCPS BATTERIES	3	4.30	12.90
GENERATOR LINE CONTACTOR (3PDT-50KVA)	2	3.30	6.60
GENERATOR LINE CONTACTOR (3PDT-7KVA)	1	0.72	0.72
EXTERNAL POWER CONTACTOR (3PDT-50KVA)	1	3.30	6.60
BATTERY CONTACTOR (3PDT-30 AMP)	1	0.60	0.60
BUS TIE CONTACTOR (SPDT-30 AMP)	1	0.60	0.60
AC CIRCUIT BREAKER (25 KVA)	2	0.18	0.36
DC CIRCUIT BREAKER (30 AMP)	5	0.12	0.60
EXTERNAL POWER CONNECTOR (50KVA)	1	2.40	2.40
EXTERNAL POWER MONITOR (50 KVA)	1	2.04	2.04
COMPONENT SUPPORTS	—	—	5.26
WIRING (FEEDER WIRES ONLY)	—	—	23.00
CONNECTORS (FEEDER CONNECTORS ONLY)	—	—	12.12
MAJOR EQUIPMENT-TOTAL WEIGHT			287.00

① SEE FIGURE 54

TABLE 42. AIRCRAFT II TOTAL WEIGHT SUBJECT TO TRADE

EQUIPMENT ITEM ①	QUANTITY PER A/C	UNIT WEIGHT (LB)	TOTAL WEIGHT (LB)
AMAD - AIRFRAME MTD. ACESSORY DRIVE	2	110.0	220.0
ATS/M - AIR TURBINE START MOTOR	2	32.0	64.0
APU/EPU - AUXILIARY/EMERGENCY POWER UNIT	1	399.0	399.0
LC - LOAD COMPRESSOR	1	75.0	75.0
PNEUMATIC DUCTING AND FITTINGS	—	—	8.4
PNEUMATIC CHECK VALVE (1½" PORT)	1	0.3	0.3
PNEUMATIC CHECK VALVE (1⅛" PORT)	2	0.2	0.4
PNEUMATIC SOLENOID SHUTOFF VALVE	4	1.2	4.8
PNEUMATIC GROUND CONNECTION	1	0.8	0.8
APU/EPU START VALVE	1	0.3	0.3
PMG - PERMANENT MAGNET GENERATOR	1	0.7	0.7
SUPPORTS AND MISC.	—	—	60.7
ELECTRICAL SYSTEM (SEE TABLE 41	—	—	287.0
HYDRAULIC SYSTEM (SEE TABLE 38			1197.6
		TOTAL WEIGHT	2319.0

① SEE FIGURE 44

5.0 TRADE STUDY RESULTS

5.1 Weight Trade - The weight summary for the two primary aircraft configurations studied are shown in table 43. It can be seen in the table that the gross takeoff weight of Aircraft I (the all electric version) was 1245 lbs. heavier than that of Aircraft II (the more conventional hydraulic-electrical configuration). From tables 27 and 42 it can be seen that the difference in the basic system weights subject to trade was 498 lbs. From this it is apparent that the growth factor for this type of airplane was 2.5.

5.2 Reliability and Maintainability Trades - The Reliability and Maintainability (R&M) trade was primarily oriented toward identifying the differences affecting the operating and support costs between the Aircraft I and II configurations. Only those major equipment items impacted by the actuation concept were identified for the R&M trades. The basic approach for the R&M trade was as follows:

1. A list of major components affected by the configuration differences was identified including actuators, electric power and generation system, and hydraulic power system.

2. For each component identified, R&M parameters based on projecting current operating data to that expected in the 1990+ time frame were estimated. Current operating data for the components used included the following sources:

   a) F-15 AFM 66-1 Maintenance data for the period October 1978 through September 1979 summarized by the Rockwell International Maintenance Analysis Model (MAM).

   b) B-1B Aircraft - "Reliability and Maintainability Allocations, Assessments, and Analysis," Rockwell International Report, NA-81-745-1, dated 2 April 1982.

   c) A-7 Aircraft - "Design Development and Evaluation of Lightweight Hydraulic System Hardware - Phase I, NADC-77108-30, North American Aircraft Division, Rockwell International Corporation, Contract N 62269-78-C-0363, 30 January 1981.

   d) Nonelectronic Reliability Notebook, Revision to Section 2, RADS-TR-69-458.

3. Estimates were made for:

   a) Mean-Time-Between-Maintenance (MTBM) including inherent failures, induced failures, and no defects resulting from a suspected failure.

   b) Mean-Time-Between-Removal (MTBR) to reflect demands on the supply system and the intermediate level maintenance shops.

TABLE 43. WEIGHT SUMMARY

	A/C I ALL ELECT		A/C II HYD.-ELECT (BASELINE)			
STRUCTURE GROUPS	(7948)		(7683)			
WING GROUP	1991		1925			
TAIL GROUP - HORIZONTAL	270		261			
- VERTICAL	302		292			
BODY GROUP	3872		3743			
ALIGHTING GEAR GROUP - MAIN	982		949			
- AUXILIARY	147		142			
- ARRESTING	91		88			
ENGINE SECTION OR NACELLE GROUP	48		46			
AIR INDUCTION SYSTEM	245		237			
PROPULSION GROUP	(4610)		(4456)			
ENGINE (AS INSTALLED)	3125		2982			
ACCESSORY GEAR BOXES & DRIVES	180		220			
EXHAUST SYSTEM						
COOLING & DRAIN PROVISIONS	33		30			
ENGINE CONTROLS	43		40			
STARTING SYSTEM	154		166			
FUEL SYSTEM	1075		1018			
FAN (AS INSTALLED)						
HOT GAS DUCT SYSTEM						
EQUIPMENT GROUPS	(7040)		(6509)			
FLIGHT CONTROLS GROUP	1655		1049			
AUXILIARY POWER PLANT GROUP	269		432			
INSTRUMENTS GROUP	170		170			
HYDRAULIC & PNEUMATIC GROUP			679			
ELECTRICAL GROUP	1131		364			
AVIONICS GROUP	2290		2290			
ARMAMENT GROUP	300		300			
FURNISHINGS AND EQUIPMENT GROUP	595		595			
AIR CONDITIONING GROUP	630		630			
ANTI-ICING GROUP						
PHOTOGRAPHIC GROUP						
LOAD & HANDLING GROUP						
DRAG CHUTE ASSY						
TOTAL WEIGHT EMPTY	(19598)		(18648)			
CREW	430		430			
FUEL - UNUSABLE	176		170			
FUEL - USABLE	11909		11623			
OIL - ENGINE	20		19			
PASSENGERS / CARGO						
ARMAMENT						
A/A MISSILES	515		515			
MISSILE LAUNCHERS	155		155			
PAYLOAD	4330		4330			
PAYLOAD FAIRINGS	51		51			
EQUIPMENT						
LIQUID NITROGEN	52		51			
PAVE SHIELD EXPENDABLES	52		51			
TOTAL USEFUL LOAD	(17690)		(17395)			
TAKEOFF GROSS WEIGHT	37288		36043			
FLIGHT DESIGN GROSS WEIGHT						
LANDING DESIGN GROSS WEIGHT						

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c) Mean-Time-to-Repair (MTTR) to reflect the average on aircraft time to repair.

d) On aircraft Maintenance-Man-Hours-Per Flight Hour (MMH/FH) to reflect an average maintenance man-hours per flight hour used to maintain the aircraft.

The above parameters were estimated for each listed component and for all the components are shown in tables 44 and 45.

Comparison of the reliability/maintainability results shown by the totals for the two aircraft configurations shows that the hydraulic Aircraft II has 11.68% improvement in the MTBM, and 18.4% improvement in the MMH/FH over the electric Aircraft I. The MTTR values are practically the same for both configurations. Evaluation of the MTBR's indicates that there are approximately five (5) times as many repairs through defective or suspected replacement on the electrical aircraft.

4. Mission Completion Success Probability (MCSP) - Aircraft I and Aircraft II system designs provide practically the same degree of redundancy for the actuation systems. Therefore, relative trends in the MCSP for the affected aircraft configurations can be established on the basis of comparison of total failure or maintenance rates estimated for all the components listed respectively for each type of aircraft.

Based on the data in Tables 44 and 45, the MCSP of the Hydraulic Aircraft II is somewhat higher than the MCSP of the electric Aircraft I.

The unreliability estimate of the electrical Aircraft I based only on the failure/maintenance rate count of the identified components is 11.68% higher than the unreliability of the 8000 psig Hydraulic Aircraft II.

5. Design Reliability-Maintainability Comments - The following summarizes some of the design features of the two proposed aircraft configurations:

   a) The 8000 psid hydraulic power generation and distribution system used smaller size components and tubing than the 3000 psig system. This feature considerably improved accessibility, and reduced maintenance times and costs as compared to the 3000 psig system.

   b) Use of smaller size tubing permitted use of coiled tubing to the exclusion of swivel joints which reduced leakage and, hence maintenance costs, and improved the reliability as compared to the 3000 psi system.

   c) In the year 1990 + on-aircraft maintenance of the electrical-mechanical drive components will be limited primarily to remove/replace activities. This would reduce flight line and increase intermediate level man-hour requirements.

TABLE 44 RELIABILITY-MAINTAINABILITY DATA
AIRCRAFT I (ELECTRIC) (SHEET 1 OF 2)

COMPONENT	QUANTITY	MDR PER 10^6 HOURS	MTBM HRS.	MTBR HRS.	MTTR HRS.	MMH PER FL-HR.
INBOARD FLAP ACT.	6	250.0	4000	4400	3.5	0.00153
MIDSPAN FLAP ACT.	6	250.0	4000	4400	3.5	0.00153
AILERON ACT.	4	331.0	3021	3300	3.5	0.00184
UPPER RUDDER ACT.	6	331	3021	3300	3.5	0.00184
LOWER RUDDER ACT.	6	331	3021	3300	3.5	0.00184
LEAD. EDGE FLAP ACT	12	331	3021	3300	3.75	0.00184
CANARD ACT.	4	120	8333	9166	3.5	0.00073
THRUST VECTOR VANE	2	120	8333	9166	3.0	0.00073
ENG. EXT. FLAP ACT.	4	120	8333	9166	3.2	0.00073
ENG. PLUG THR. ACT.	2	240	4167	4584	3.2	0.00147
ENG. THRUST REV. ACT.	2	180	5556	6110	3.7	0.00110
NOSE L. G. ACT.	1	120	8333	9166	3.2	0.00073
MAIN L. G. ACT	2	120	8333	9166	3.7	0.00073
NOSE GEAR STEER.	1	148.5	6734	7407	3.2	0.00091
M. G. BRAKES	2	148.5	6734	7407	3.5	0.00091
ECS RH RAMAIR	1	120	8333	9166	3.5	0.00073
ECS LH RAMAIR	1	120	8333	9166	3.5	0.00073
ECS EXCH. DOOR E. B.	1	120	8333	9166	3.2	0.00073

TABLE 44 RELIABILITY-MAINTAINABILITY DATA
AIRCRAFT I (ELECTRIC) SHEET 2 OF 2)

COMPONENT	QUANT.	MDR PER 10^6 HOURS	MTBM HRS.	MTBR HRS.	MTTR HRS.	MMH PER FL-HR
ARMAMENT-ACT	1	90.9	11,000	12,100	3.7	0.00056
REFUEL RECEPT. MOTOR	1	90.9	11,000	12,100	3.25	0.00056
GENERATOR-270VDC	4	489.7	2040	5250	3.25	0.00300
DRAIN+FILL CON VERT.	2	40.0	50,000	55,000	3.0	0.00012
TRANSF-RECTIF.	4	11.5	86,956	250,000	3.0	0.00070
CYCLOCONVERTER	4	40.0	25,000	27,500	3.0	0.00024
GENERATOR-CONTROL	4	89.94	11,119	125,000	3.0	0.00055
BATTERY-270VDC	1	348.8	2867	8768	3.0	0.00213
APU MOTOR	1	148.5	6734	7407	3.25	0.00091
APU GENERATOR	2	326.4	3063	9370	3.25	0.00200
APU TRANS-RECT.	2	11.5	86,956	250,000	3.0	0.00006
APU CYCLOCONV.	2	20.0	50,000	55,000	3.0	0.00012
INVERT. (START)	2	20.0	50,000	150,000	2.5	0.00011
REVERSIBLE SCR	2	12.54	79,745	87,720	2.0	0.00007
START RELAYS	10	6.27	159,490	175,439	2.5	0.00003
TOTAL	105	21,155	47.27	63.38	3.25	0.11681

TABLE 45 RELIABILITY-MAINTAINABILITY DATA
AIRCRAFT II (HYDRAULIC) (SHEET 1 OF 2)

COMPONENT	QUANT.	MDR PER 10^6 HOURS	MTBM HRS.	MTBR HRS.	MTTR HRS.	MMH PER FL-HR
INB. FLAP ACT	6	289.6	3452	23014	3.75	0.00153
MIDSPAN FLAP ACT	6	257.4	3884	25893	3.75	0.00136
AILERON ACT	4	231.7	4315	28767	3.75	0.00122
UPPER RUDDER ACT	6	231.7	4315	28767	3.75	0.00122
LOWER RUDDER ACT	6	231.7	4315	28767	3.75	0.00122
L. E. FLAP ACT.	12	231.7	4315	28767	3.75	0.00122
CANARD ACT.	4	148.3	6743	53944	3.5	0.00078
T. V. VANE ACT.	2	148.3	6743	53944	3.5	0.00078
ENG. EXT. FLAP ACT	4	185.3	5394	43152	3.5	0.00098
ENG. PLUG THR. ACT	2	331.1	3020	20,133	3.5	0.00175
ENG. THRUST REV.	2	231.7	4315	28767	3.0	0.00122
NOSE L. G. ACT	1	170.9	5850	46800	3.75	0.00090
MAIN L. G. ACT	2	206.6	4838	38704	3.00	0.00109
NOSE GEAR STEER	1	231.6	4316	28773	3.5	0.00122
MAIN GEAR BRAKES	2	257.4	3884	25893	3.5	0.00136
ECS RH RAMAIR	1	74.1	13486	107,888	3.50	0.00039
ECS LH RAMAIR	1	74.1	13486	107,888	3.50	0.00039
ECS EXCH. DR ACT.	1	148.3	6743	53,944	3.75	0.00078

TABLE 45 RELIABILITY-MAINTAINABILITY DATA
AIRCRAFT II (HYDRAULIC) (SHEET 2 OF 2)

COMPONENT	QUANT.	MDR PER 10^6 HOURS	MTBM HRS.	MTBR HRS.	MTTR HRS.	MMH PER FL-HR
ARMAMENT-ACT	1	231.7	4315	28767	3.0	0.00122
REFUEL RECEP. MOTOR	1	115.2	8680	37200	3.5	0.00061
PRIM. PUMP	4	356.2	2807	12,047	3.5	0.00188
APU PUMP	1	296.9	3368	14455	3.5	0.00157
PRIMARY RESERV.	2	30.9	32,320	300,000	3.0	0.00016
EMERG. RESERV.	1	30.9	32,320	300,000	3.0	0.00016
ACCUMULATOR	1	83.3	12,000	120,000	3.5	0.00044
BRAKE RESERV.	2	30.9	32,300	300,000	3.5	0.00016
GENERATOR 40 KVA	2	489.7	2042	6250	3.25	0.00259
GENERATOR 10 KVA	1	326.4	3063	9370	3.25	0.00172
TRANSF/RECTIF.	2	11.5	86956	250,000	3.0	0.00006
INVERTER	1	20.0	50000	150,000	2.5	0.00011
VOLT. REGUL.	1	89.3	11,200	33600	2.0	0.00047
BATTERY	2	348.8	2867	8760	2.0	0.00184
POWER CONTAC.	12	9.5	105,241	210,000	2.0	0.00005
TOTAL	97	18.942.9	52.7	303.08	3.31	0.09866

d) Fly-by-wire signal transmission in conjunction with power-by-wire required exceptionally high electrical power reliability for future airborne all-electrical power systems.

e) To meet the failure and reliability requirement of the 8000 psig hydraulic Aircraft II, the electric Aircraft I had to provide at least 3 completely independent dedicated power systems to match redundancy in Aircraft II.

f) The actuators of Table 45 (all-electric) included the reliability impacts of their inverters which largely accounted for their deficient reliability with respect to the actuators of Table 5.3 (hydraulic-electric).

5.3 Life Cycle Costs - Life cycle cost estimates were developed for each of the "Airplanes" defined. The baseline aircraft (Aircraft II) was an electric-hydraulic powered aircraft, and the alternate configuration was an "all electric" approach (Aircraft I).

5.3.1 Methodology - The estimates were developed utilizing the Integrated Aircraft Life Cycle Cost Model II (IALCCM II). IALCCM II was a computer program developed by Rockwell in support of previous advanced tactical fighter studies which estimated RDT&E, Production, and Operations and Support Cost. This model provided preliminary cost data during conceptual and preliminary design states.

5.3.2 Ground Rules - The ground rules for developing the cost estimates were as follows:

1. Number of flight test aircraft (10)
2. Number of production aircraft (500)
3. Number of aircraft in the field (432)
4. O&S cost for field aircraft
    a. 10 year operational life
    b. 5 year buildup
    c. 24 aircraft per squadron
    d. 25 flight hours/aircraft/month
    e. Assumed fuel cost $1.26/U.S. dollars (1982 dollars)
5. LCC was submitted in constant 1982 U.S. dollars and 1995 dollars. (The 1995 dollars were developed utilizing inflation factors from the USAF Cost and Planning Factor Manual [AF Regulation 173-13]).

5.3.3 Cost Summaries - Cost data were developed for each of the two configurations. LCC summaries are provided in tables 46 and 47 in 1982 dollars, and table 48 and 49 in 1995 dollars.

TABLE 46 AIRCRAFT I LIFE CYCLE COSTS - 1982 DOLLARS

AIRCRAFT:ATSI ACTUATION TRADE STUDY ALL ELECTRIC
COSTS IN FY 1982 MILLION DOLLARS

LIFE CYCLE COST	16096.95	TOTAL PRODUCTION AIRCRAFT	500
		TOTAL UE	432
		COMMAND SUPPORT	42
RDT&E	1614.16	ATTRITION	26
AIRFRAME	1012.53		
PROPULSION	544.24	TOTAL PROTOTYPE AIRCRAFT	10
AVIONICS	57.40		
OTHER	0.0		
ACQUISITION	6373.10		
PRODUCTION	5985.89		
FLYAWAY	5347.24	UNIT AVERAGE FLYAWAY COST	10.694
INITIAL SPARES	339.81	AIRFRAME FLYAWAY	7.210
INITIAL CSE	170.52	PROPULSION FLYAWAY	2.162
TRAINING EQUIPMENT	69.51	AVIONICS FLYAWAY	1.322
TECHNICAL DATA	58.82	OTHER FLYAWAY	0.0
OTHER INVESTMENT	387.21		
TRANSPORTATION	119.72		
INITIAL PERSONNEL ACQUISITION	105.96		
INITIAL PERSONNEL TRAINING	161.53		
FACILITIES	0.0		
TOTAL OPERATIONS FOR 10 YEARS	8109.70	OPERATIONS DATA:	
RECURRING INVESTMENT & MISC. LOGISTICS	5540.43	UE PER SQUADRON	24.0
COMMON CSF	516.28	UTIL RATE FHRS/UE/MONTH	25.0
AVIATION FUEL @ 1.26/GAL	1671.30	CREW RATIO	1.1
BASE LEVEL MAINTENANCE MATERIAL	365.44	PILOTS/CREW	1.0
DEPOT LEVEL MAINTENANCE	2114.13	OTHER OFFICERS/CREW	0.0
CLASS IV MODIFICATIONS	207.62	MAINTENANCE MAN HOURS/FHR	28.3
TRAINING MUNITIONS	201.85	MUNIT MAINT MEN/UE	6.0
REPLENISHMENT SPARES	455.16	FUEL FLOW GPH	1024.0
VEHICULAR EQUIPMENT	8.65	REPL SPARES $/FHR	351.2
PAY AND ALLOWANCES	2014.05	BASE MAINT MTL $/FHR	282.0
MFP- BOS/RPM SUPPORT	132.35	DEPOT MAINT $/FHR	1460.0
MEDICAL SUPPORT	90.18	DEPOT MAINT $/UE/YR	51387.4
PERSONNEL SUPPORT (PCS MOVES)	64.54	COMMON OSE $/UE/YR	119508.3
PERSONNEL ACQUISITION AND TRAINING.	268.15		

TABLE 47 AIRCRAFT II LIFE CYCLE COSTS - 1982 DOLLARS

AIRCRAFT:ATSII ACTUATION TRADE STUDY BASELINE
COSTS IN FY 1982 MILLION DOLLARS

LIFE CYCLE COST	12313.76	TOTAL PRODUCTION AIRCRAFT	500
		TOTAL UE	432
		COMMAND SUPPORT	42
RDT&E	1437.60	ATTRITION	26
AIRFRAME	846.04		
PROPULSION	534.16	TOTAL PROTOTYPE AIRCRAFT	10
AVIONICS	57.40		
OTHER	0.0		
ACQUISITION	5674.29		
PRODUCTION	5307.36		
FLYAWAY	4741.10	UNIT AVERAGE FLYAWAY COST	9.482
INITIAL SPARES	301.29	AIRFRAME FLYAWAY	6.088
INITIAL OSE	151.19	PROPULSION FLYAWAY	2.072
TRAINING EQUIPMENT	61.63	AVIONICS FLYAWAY	1.322
TECHNICAL DATA	52.15	OTHER FLYAWAY	0.0
OTHER INVESTMENT	366.93		
TRANSPORTATION	106.15		
INITIAL PERSONNEL ACQUISITION	102.80		
INITIAL PERSONNEL TRAINING	157.98		
FACILITIES	0.0		
TOTAL OPERATIONS FOR 10 YEARS	5701.88	OPERATIONS DATA:	
RECURRING INVESTMENT & MISC. LOGISTICS	3226.39	UE PER SQUADRON	24.0
COMMON OSE	199.72	UTIL RATE FHRS/UE/MONTH	25.0
AVIATION FUEL @ 1.26/GAL	1623.92	CREW RATIO	1.1
BASE LEVEL MAINTENANCE MATERIAL	122.97	PILOTS/CREW	1.0
DEPOT LEVEL MAINTENANCE	459.85	OTHER OFFICERS/CREW	0.0
CLASS IV MODIFICATIONS	184.09	MAINTENANCE MAN HOURS/FHR	26.3
TRAINING MUNITIONS	201.85	MUNIT MAINT MEN/UE	6.0
REPLENISHMENT SPARES	425.69	FUEL FLOW GPH	994.9
VEHICULAR EQUIPMENT	8.30	REPL SPARES $/FHR	328.5
PAY AND ALLOWANCES	1940.18	BASE MAINT MTL $/FHR	94.9
MFP- 803/RPM SUPPORT	127.15	DEPOT MAINT $/FHR	199.9
MEDICAL SUPPORT	86.56	DEPOT MAINT $/UE/YR	46464.3
PERSONNEL SUPPORT (PCS MOVES)	62.10	COMMON OSE $/UE/YR	46231.7
PERSONNEL ACQUISITION AND TRAINING	259.61		

TABLE 48 AIRCRAFT I LIFE CYCLE COSTS - 1995 DOLLARS

AIRCRAFT:ATSI ACTUATION TRADE STUDY ALL ELECTRIC
COSTS IN FY 1995 MILLION DOLLARS

LIFE CYCLE COST	31107.82	TOTAL PRODUCTION AIRCRAFT	500
		TOTAL UE	432
		COMMAND SUPPORT	42
RDTE	3107.83	ATTRITION	26
AIRFRAME	1984.02		
PROPULSION	1066.41	TOTAL PROTOTYPE AIRCRAFT	10
AVIONICS	57.40		
OTHER	0.0		
ACQUISITION	11764.29		
PRODUCTION	11019.74		
FLYAWAY	9844.00	UNIT AVERAGE FLYAWAY COST	19.688
INITIAL SPARES	625.56	AIRFRAME FLYAWAY	14.129
INITIAL OSE	313.91	PROPULSION FLYAWAY	4.237
TRAINING EQUIPMENT	127.97	AVIONICS FLYAWAY	1.322
TECHNICAL DATA	108.28	OTHER FLYAWAY	0.0
OTHER INVESTMENT	744.55		
TRANSPORTATION	225.39		
INITIAL PERSONNEL ACQUISITION	207.63		
INITIAL PERSONNEL TRAINING	310.53		
FACILITIES	0.0		
TOTAL OPERATIONS FOR 10 YEARS	16235.70	OPERATIONS DATA:	
RECURRING INVESTMENT & MISC. LOGISTICS	10667.27	UE PER SQUADRON	24.0
COMMON OSE	959.07	UTIL RATE FHRS/UE/MONTH	25.0
AVIATION FUEL $ 2.47/GAL	3272.63	CREW RATIO	1.1
BASE LEVEL MAINTENANCE MATERIAL	710.10	PILOTS/CREW	1.0
DEPOT LEVEL MAINTENANCE	4069.31	OTHER OFFICERS/CREW	0.0
CLASS IV MODIFICATIONS	502.22	MAINTENANCE MAN HOURS/FHR	28.3
TRAINING MUNITIONS	395.54	MUNIT MAINT MEN/UE	6.0
REPLENISHMENT SPARES	661.47	FUEL FLOW GPH	1024.0
VEHICULAR EQUIPMENT	16.94	REPL SPARES $/FHR	664.7
PAY AND ALLOWANCES	4460.43	BASE MAINT MTL $/FHR	547.9
MFP- SOS/RPM SUPPORT	259.35	DEPOT MAINT $/FHR	2804.2
MEDICAL SUPPORT	170.71	DEPOT MAINT $/UE/YR	160696.9
PERSONNEL SUPPORT (PCS MOVES)	120.48	COMMON USE $/UE/YR	222006.3
PERSONNEL ACQUISITION AND TRAINING	525.46		

TABLE 49 AIRCRAFT II LIFE CYCLE COSTS - 1995 DOLLARS

AIRCRAFT:ATSII ACTUATION TRADE STUDY BASELINE
COSTS IN FY 1995 MILLION DOLLARS

LIFE CYCLE COST	24721.02
RDT&E	2615.73
AIRFRAME	1511.66
PROPULSION	1046.67
AVIONICS	57.40
OTHER	0.0
ACQUISITION	10499.92
PRODUCTION	9793.05
FLYAWAY	8748.19
INITIAL SPARES	555.91
INITIAL OSE	278.97
TRAINING EQUIPMENT	113.73
TECHNICAL DATA	96.23
OTHER INVESTMENT	706.88
TRANSPORTATION	195.86
INITIAL PERSONNEL ACQUISITION	201.45
INITIAL PERSONNEL TRAINING	309.57
FACILITIES	0.0
TOTAL OPERATIONS FOR 10 YEARS	11605.37
RECURRING INVESTMENT & MISC. LOGISTICS	6240.02
COMMON CSE	371.74
AVIATION FUEL @ 2.47/GAL	3179.65
BASE LEVEL MAINTENANCE MATERIAL	239.02
DEPOT LEVEL MAINTENANCE	891.42
CLASS IV MODIFICATIONS	339.68
TRAINING MUNITIONS	395.54
REPLENISHMENT SPARES	806.73
VEHICULAR EQUIPMENT	16.26
PAY AND ALLOWANCES	4316.17
MFP- ROS/FPM SUPPORT	249.15
MEDICAL SUPPORT	169.62
PERSONNEL SUPPORT (PCS MOVES)	121.69
PERSONNEL ACQUISITION AND TRAINING	508.72

TOTAL PRODUCTION AIRCRAFT	500
TOTAL UE	432
COMMAND SUPPORT	42
ATTRITION	26
TOTAL PROTOTYPE AIRCRAFT	10
UNIT AVERAGE FLYAWAY COST	17.496
AIRFRAME FLYAWAY	12.114
PROPULSION FLYAWAY	4.061
AVIONICS FLYAWAY	1.322
OTHER FLYAWAY	0.0
OPERATIONS DATA:	
UE PER SQUADRON	24.0
UTIL RATE FHRS/UE/MONTH	25.0
CREW RATIO	1.1
PILOTS/CREW	1.0
OTHER OFFICERS/CREW	0.0
MAINTENANCE MAN HOURS/FHR	26.3
MUNIT MAINT MEN/UE	6.0
FUEL FLOW GPH	994.9
REPL SPARES $/FHR	622.5
BASE MAINT MTL $/FHR	184.4
DEPOT MAINT $/FHR	384.3
DEPOT MAINT $/UE/YR	91049.7
COMMON CSE $/UE/YR	86051.9

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APPENDIX A

AIRPLANE ACTUATION TRADE STUDY

This appendix includes the initial work done by AiResearch under the direction of S. Rowe, as part of their effort in accordance with service agreement L9FM011231-405 to define the size, weight, and performance characteristics of the inboard flap actuator. With further refinements, this led to the final definition of the "hingeline installation" shown in Figure 25 as well as the applicable entries in Tables 18 through 20.

ALL ELECTRIC AIRPLANE STUDY

Appendix A

DATE [illegible]
PART NO. [illegible]
PREPARED BY [illegible]

CALC. NO. [illegible] SHEET NO. 1
MODEL NO.

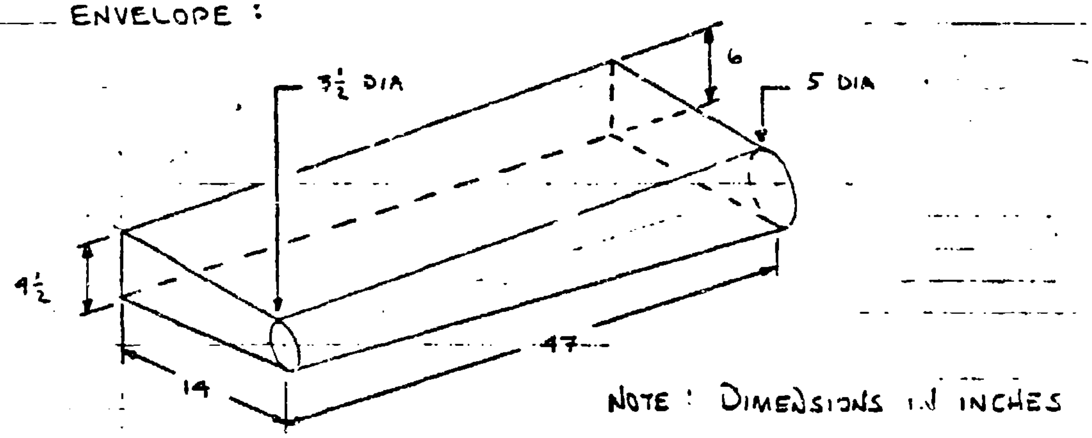
APPENDIX A

INBOARD FLAP PRELIMINARY DESIGN

SIZE THREE ACTUATORS, EACH HAVING THE FOLLOWING PERFORMANCE:

NO-LOAD RATE	100 DPS
DESIGN-LOAD RATE	50 DPS
DESIGN-LOAD	162 K-IN-LB
DESIGN-LOAD & RATE POWER	21.4 HP
STALL-LOAD	216 K-IN-LB
BANDWIDTH ($A = \pm 2.5°$, $J_L = 50$ IN-LB-SEC^2)	
• AR	3 HZ $^{+2}_{-3}$ DB
• ϕ	-30° MAX
STIFFNESS	1.82×10^6 IN-LB-RAD^{-1}

ALL THREE ACTUATORS MUST FIT THE FOLLOWING ENVELOPE:



NOTE: DIMENSIONS IN INCHES

DATE E-23-80 CALC. NO. 1-001 SHEET NO. 2

PART NO. RAATS MODEL NO.

PREPARED BY S ROWE *APPENDIX A*

MOTOR

ASSUME AN OVERALL DRIVE EFFICIENCY OF

$$\eta = 0.85$$

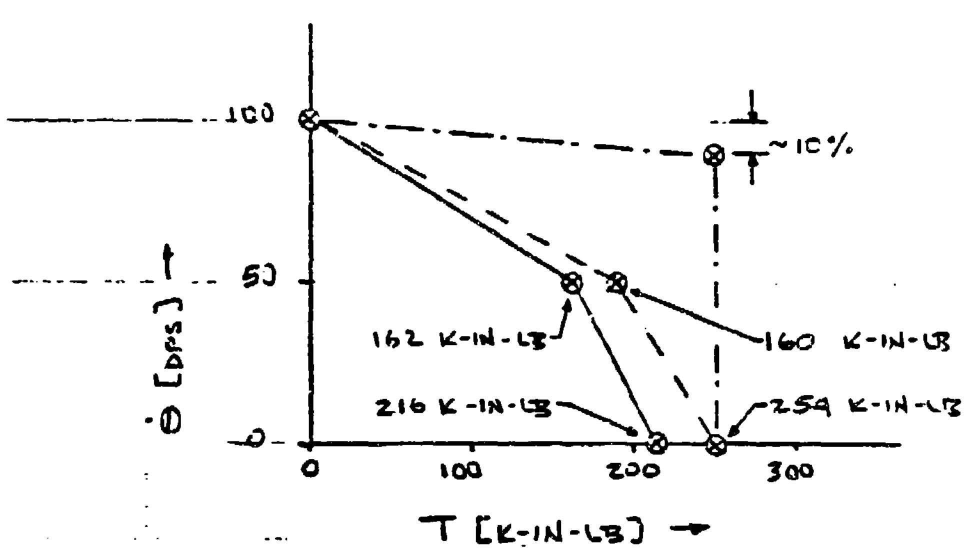
INPUT POWER REQUIREMENT IS

$$\dot{W}_{IN} = \dot{W}_{OUT} \div \eta$$

$$= 21.4 \div 0.85$$

$$= 25.2 \text{ HP}$$

THE ACTUATOR MUST HAVE THE FOLLOWING SPEED-TORQUE CHARACTERISTICS (SOLID PLOT):



NOTES: ACTUATOR (WITH LOSSES)
ACTUATOR (WITHOUT LOSSES)
MOTOR

DATE 8-23-80 CALC. NO. 1-001 SHEET NO. 3

PART NO. MODEL NO.

PREPARED BY S. ROWE

APPENDIX A

SELECT A MOTOR WITH THE SPEED-TORQUE CHARACTERISTICS SHOWN IN THE FIGURE (10% SPEED DROOP). THE MOTOR POWER RATING IS THEN:

$$\dot{W}_M = \dot{\Theta}_{NL} \times 90\% \times T_{IN}$$

$$= (100 \div 57.29) \times (90 \div 100) \times (254)$$

$$\doteq 399 \text{ K-IN-LB-SEC}^{-1}$$

$$\doteq 60.5 \text{ HP}$$

SO SELECT A MOTOR WITH A POWER RATING OF

$$\dot{W}_M = 60 \text{ HP}$$

NEXT DETERMINE TIME CONSTANT* . EVALUATE PERFORMANCE WITH FOLLOWING MOTOR [1]:

τ_M	100 MSEC
$\dot{W}_M$	60 HP
$\dot{\Theta}_{M-NL}$	24.2 KRPM
$\dot{\Theta}_{M-MX}$	21.0 KRPM
T_{M-MX}	181.9 IN-LB
J_M	8.092×10^{-3} IN-LB-SEC^2

USE TI-59 PROGRAM 'RAATS' TO EVALUATE ACTUATOR PERFORMANCE. PROGRAM OUTPUT IS SHOWN BELOW (SEE APPENDIX FOR PROGRAM NOMENCLATURE) [2].

* TIME TO REACH $\dot{\Theta}_{M-MX}$ FOR T_{M-MX} CONSTANT

DATE 8-13-80
PART NO. RAATS
PREPARED BY S. Rowe

CALC. NO. 1-001 SHEET NO. 4
MODEL NO.
APPENDIX A

A = ±1°

```
RAATS

      IN:
    24200.      OMNL
    21000.      OMMX
     181.9      TMMX
  0.008092        JM
        1.        NM
      100.      OANL
       50.        JL
      0.85       NGR
    0.0174         A

     OUT:
      100.      OANL
86.7768595      OAMX
 224500.98      TAMX
29.77714484       WA
     1452.        GR
```

A = ±0.5°

```
RAATS

      IN:
    24200.      OMNL
    21000.      OMMX
     181.9      TMMX
  0.008092        JM
        1.        NM
      100.      OANL
       50.        JL
      0.85       NGR
    0.0087         A

     OUT:
      100.      OANL
86.7768595      OAMX
 224500.98      TAMX
42.11124208       WA
     1452.        GR
```

A = ±0.25°

```
RAATS

      IN:
    24200.      OMNL
    21000.      OMMX
     181.9      TMMX
  0.008092        JM
        1.        NM
      100.      OANL
       50.        JL
      0.85       NGR
   0.00435         A

     OUT:
      100.      OANL
86.7768595      OAMX
 224500.98      TAMX
59.55428968       WA
     1452.        GR
```

MOTOR EVALUATION: τ_M = 100 msec, $\dot{W}_M$ = 60 HP

REPORT
PAGE

DATE 5-23-8? CALC. NO. 1-031 SHEET NO. 5

PART NO. RAATS MODEL NO.

PREPARED BY S ROWE APPENDIX A

ACTUATOR PERFORMANCE IS ACCEPTABLE SO USE SUGGESTED MOTOR. MOTOR DATA IS SUMMARIZED BELOW:

TIME CONSTANT	100 MSEC
POWER RATING	60 HP
NO-LOAD SPEED	24.2 KRPM
MAX TORQUE	181.9 IN-LB
CURRENT LIMIT	208.2 AMP
DIAMETER	3.7 IN
LENGTH	10.5 IN
WEIGHT	13.7 LB (USE 14 LB)

CLASSIFICATION

REPORT

PAGE

DATE 6-25-80 CALC. NO. 1-331 SHEET NO. [illegible]

PART NO. RAATS MODEL NO.

PREPARED BY S. ROWE [illegible] APPENDIX A

HINGELINE DRIVE

USE DIAMETER OF

$D = 3.5$ IN

FROM PARAMETRIC DATA [3, 4, 5] USING THE CURVES MARKED 'ULTIMATE' AND 'LUG'

$\bar{T} = 3.6 \times 10^4$ IN-LB-IN^{-1}

$\bar{K} = 5.6 \times 10^5$ IN-LB-RAD^{-1}-IN^{-1}

$\bar{W} = 1.4$ LB-IN^{-1}

SINCE DATA IS FOR 6 PLANET DRIVES, INCREASE PARAMETERS FOR CASE OF 12 PLANETS:

$\bar{T} = 3.6 \times 10^4$ IN-LB-IN^{-1} (+ 0%)

$\bar{K} = 1.1 \times 10^6$ IN-LB-RAD^{-1}-IN^{-1} (+ 100%)

$\bar{W} = 2.1$ LB-IN^{-1} (+ 50%)

DETERMINE LENGTH OF DRIVE TO ACCOMMODATE ACTUATOR STALL:

$L = T_{STALL} \div \bar{T}$

$= 216 \times 10^3 \div 3.6 \times 10^4$

$= 6.0$ IN

CHECK STIFFNESS:

$K = L \times \bar{K}$

$= 6 \times 1.1 \times 10^6$ IN-LB-RAD^{-1}-IN^{-1}

$= 6.6 \times 10^6$ IN-LB-RAD^{-1}

REPORT

PAGE

DATE 5-23-80 | CALC. NO. 1-001 | SHEET NO. 7
PART NO. RAATS | MODEL NO.
PREPARED BY S. ROWE | APPENDIX A

THE DESIGN IS ACCEPTABLE, SO USE THE PROPOSED HINGELINE DRIVE. DATA IS GIVEN BELOW:

REDUCTION (EST)	10-20:1
EFFICIENCY (EST)	90-80 %
DIAMETER	3.5 IN
LENGTH	6.0 IN
WEIGHT	12.6 LB (USE 13 LB)
STIFFNESS	6.6×10^6 IN-LB-RAD^{-1}
CONFIGURATION	COMPOUND EPICYCLIC, 12 PLANET/SLICE

DATE 5-23-80 CALC. NO. 1-031 SHEET NO. 8

PART NO. TRAATS MODEL NO.

PREPARED BY S. Rowe

APPENDIX A

GEARHEAD

SCALE GEARHEAD FROM EXISTING PDU [6]. RETAIN 90° INPUT FOR MOTOR. APPROXIMATE DIMENSIONS AND DATA FOR THE UNIT ARE:

REDUCTION	1 - 150 : 1
EFFICIENCY	94 %
DIAMETER	4.8 IN
LENGTH	5.0 IN
WEIGHT	~ 11.6 LB

MAXIMUM INPUT TORQUE FOR THE UNIT IS

$$T_{MAX} = 363 \text{ IN-LB}$$

USING CUBE ROOT RULE TO SCALE:

$$SF^{*} = \left(\frac{T_{NEW}}{T_{OLD}}\right)^{1/3}$$

$$= \left(\frac{181.9}{363}\right)^{1/3}$$

$$= 0.794$$

NEW DIMENSIONS ARE CALCULATED AS

NEW = OLD × SF

* SF ≜ SCALE FACTOR

DATE 5-23-80 CALC. NO. 1-321 SHEET NO. 9

PART NO. RAA-C MODEL NO.

PREPARED BY S. ROWE *APPENDIX A*

THUS

$$D = (4.8) \times (0.794)$$

$$\doteq 3.8 \text{ IN}$$

$$L = (5.0) \times (0.794)$$

$$\doteq 4.0 \text{ IN}$$

ASSUMING A CONSTANT DENSITY OF

$$\rho = 0.128 \text{ LB-IN}^{-3}$$

THE WEIGHT BECOMES

$$W = V \times \rho$$

$$= \frac{D^2}{4} \pi L \rho$$

$$= \frac{(3.8)^2}{4} \pi (4.0)(0.128)$$

$$= 5.8 \text{ LB}$$

FROM THE DIMENSIONS, IT APPEARS THAT THE GEARHEAD WOULD BE PRACTICAL FOR HINGELINE APPLICATION NEARLY THE FULL LENGTH OF THE ENVELOPE. THUS USE THE SCALED GEARHEAD:

EFFICIENCY	94%
REDUCTION	1-150:1
DIAMETER	3.8 IN.
LENGTH	4.0 IN

DATE 8-23-90

PART NO. EAATS

PREPARED BY S. ROWE

CALC. NO. 1-001 SHEET NO. 12

MODEL NO.

APPENDIX A

WEIGHT 5.8 LB (USE 6 LB)

CONFIGURATION SIMPLE PLANETARY, WITH RIGHT ANGLE INPUT

DATE 5-23-80
PART NO. RAATS
PREPARED BY S. ROWE

CALC. NO. 1-021 SHEET NO. 11
MODEL NO.
APPENDIX A

INSTALLATION

USING DIMENSIONS AND WEIGHTS FROM PREVIOUS CALCULATIONS, SHOW INSTALLATION DIMENSIONS AND WEIGHTS.

THE INSTALLATION IS SHOWN IN THE FIGURE ON THE FOLLOWING PAGE. THE FOLLOWING NOTES APPLY:

(1) A CLUTCH AS BEEN INCORPORATED FOR MOTOR DISENGAGEMENT. DIMENSIONS AND WEIGHT ARE

$D = 3.7$ IN

$L = 1.0$ IN

$W = 1$ LB

(2) THE DRAWING IS APPROXIMATELY TO SCALE. IT IS NOTED THAT THE UNITS SLIGHTLY EXCEED THE LONGITUDINAL DIMENSION.

(3) ONE INVERTER PER MOTOR WILL BE REQUIRED. DATA FOR A TYPICAL INVERTER ARE [1]:

i_{LIM}	200 AMP (+ 10% TRANSIENT)
D	7.2 IN
L	18.5 IN
W	48 LB
TYPE	PASSIVE FREON, FUEL COOLED

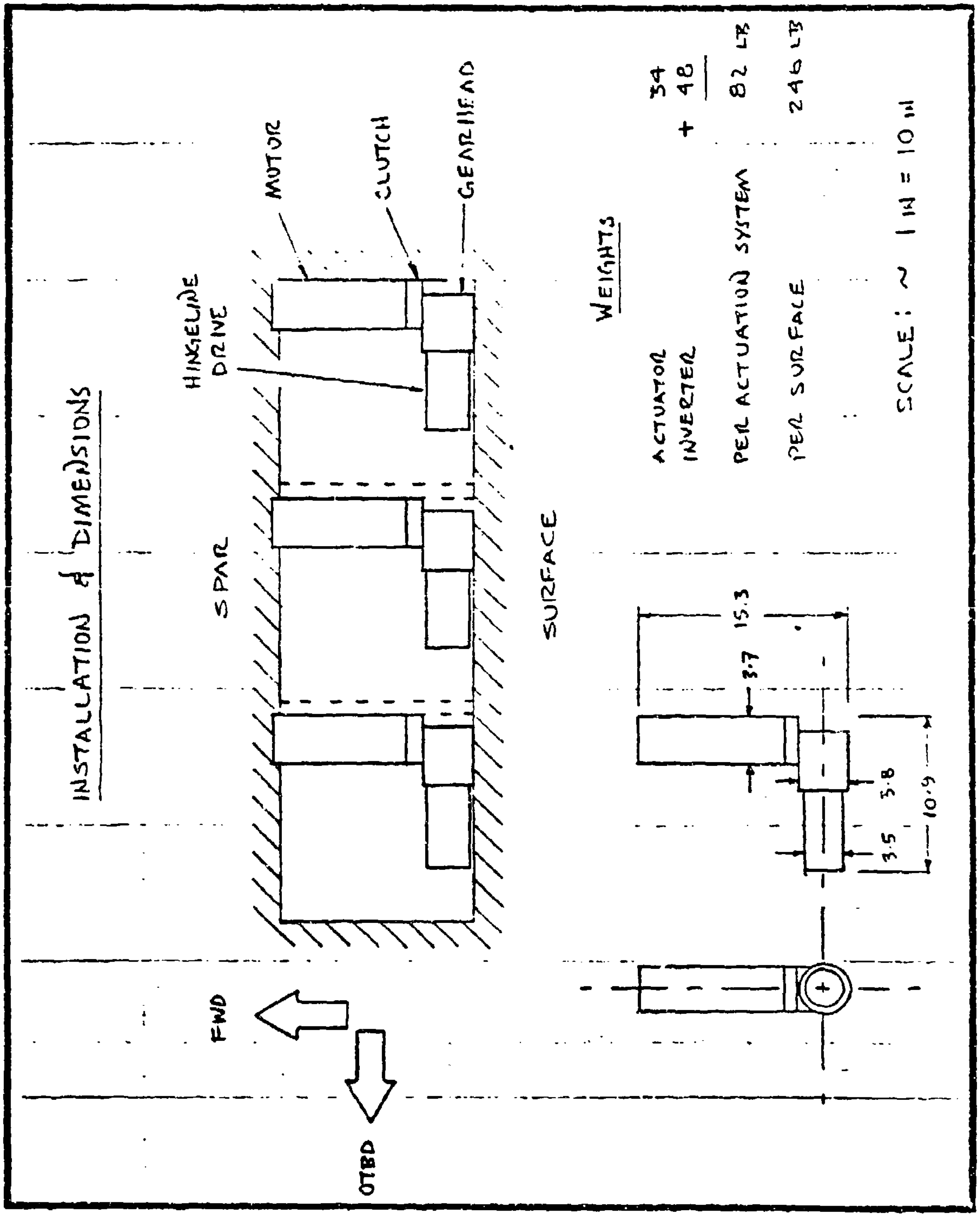
REPORT
PAGE

DATE 5-23-80 CALC. NO. 1-001 SHEET NO. 12

PART NO. MODEL NO.

PREPARED BY S. ROWE

APPENDIX A



DATE 8-23-80

PART NO.

PREPARED BY S. ROWE

CALC. NO. 1-001 SHEET NO. 13

MODEL NO.

APPENDIX A

PERFORMANCE EVALUATION

ESTIMATE ACTUATION SYSTEM PERFORMANCE FOR BEST AND WORST DRIVE EFFICIENCIES,

$$\eta_{BEST} = (0.94) \times (0.90)$$
$$= 0.846$$

$$\eta_{WORST} = (0.94) \times (0.80)$$
$$= 0.752$$

PRINTOUTS FROM TI-59 PROGRAM 'RAATS' ARE SHOWN ON THE FOLLOWING PAGE.

CALC. NO. 1-901 SHEET NO. 14

MODEL NO. APPENDIX A

RAATS

S. ROWE

$\eta = 0.846$

```
RAATS

      .IN:
    24200.      DMNL
    21000.      DMMX
     181.9      TMMX
  0.008092        JM
        1.        NM
      100.      DANL
       50.        JL
     0.846       NGR
    0.0174         A

      OUT:
      100.      DANL
86.7768595      DAMX
223444.5048     TAMX
29.77690296       WA
     1452.        GR
```

$\eta = 0.752$

```
RAATS

       IN:
    24200.      DMNL
    21000.      DMMX
     181.9      TMMX
  0.008092        JM
        1.        NM
      100.      DANL
       50.        JL
     0.752       NGR
    0.0174         A

      OUT:
      100.      DANL
86.7768595      DAMX
198617.3376     TAMX
29.77048011       WA
     1452.        GR
```

INBOARD FLAP PERFORMANCE EVALUATION

DATE 5-23-80 | CALC. NO. 1-001 | SHEET NO. 15

PART NO. RAATS | MODEL NO.

PREPARED BY S. ROWE | APPENDIX A

REFERENCES

1. AIR REPORT NO. 80-17259,
   AIRPLANE ACTUATION TRADE STUDY: FINAL REPORT

2. TI-59 PROGRAM LISTING,
   "RAATS: ROCKWELL AIRPLANE ACTUATION TRADE STUDY"

3. AIR DOCUMENT NO. S-33199,
   "ROTORY ACTUATOR PARAMETRIC DESIGN WEIGHT AND CONFIGURATION"

4. AIR DOCUMENT NO. S-33200,
   "ROTORY ACTUATOR PARAMETRIC DESIGN SPRING RATE"

5. AIR REPORT NO. 78-15310,
   PRELIMINARY DESIGN MANUAL FOR ROTORY ACTUATORS

6. AIR DRAWING NO. 2022646

DATE 8-23-80

PART NO. RAATS

PREPARED BY S. ROWE

CALC. NO. 1-001 SHEET NO. A-1

MODEL NO.

APPENDIX A

APPENDIX: TI-59 PROGRAM 'RAATS' NOMENCLATURE

NOMENCLATURE

OMNL : MOTOR NO-LOAD SPEED [RPM]

OMNX : MOTOR MAX-LOAD SPEED [RPM]

TMMX : MOTOR MAX-LOAD [IN-LB]

JM : MOTOR INERTIA [IN-LB-SEC²]

NM : NO. OF MOTORS [-]

OANL : ACTUATOR NO-LOAD SPEED [DPS]

JL : LOAD INERTIA [IN-LB-SEC²]

NGR : GEARING EFFICIENCY [-]

A : OUTPUT AMPLITUDE [RAD]

OANL : ACTUATOR NO-LOAD SPEED [DPS]

OAMX : ACTUATOR MAX-LOAD SPEED [DPS]

TAMX : ACTUATOR MAX-LOAD [IN-LB]

WA : ACTUATOR BANDWIDTH [RPS]

GR : GEAR RATIO [-]

APPENDIX B

AIRPLANE ACTUATION TRADE STUDY

This appendix includes the initial work done by AiResearch, under the direction of S. Rowe, as part of their effort under service agreement L9FM-11231-405 to define the functional characteristics of the inverters and controllers needed for implementing the "all electric" (Aircraft I) approach to the trade study.

ALL ELECTRIC AIRPLANE STUDY

SERVICE AGREEMENT 69FM-11231-405

PROGRESS REPORT FOR THE MONTH OF SEPTEMBER:

ADVANCED ELECTROMECHANICAL ACTUATION SYSTEMS

0. PREFACE

This document is submitted in conjunction with Service Agreement L9FM-11231-405, all Electric Airplane Study. The data submitted herein is presented to facilitate completion of the subject study. This document, in addition to previously transmitted data, shall serve as a progress report for the month of September 1980.

1. INTRODUCTION

Electromechanical actuation systems (EMAS) are finding increased potential for use in aircraft flight control systems (FCS) [1, 2, 3, 4]. Most advanced EMAS utilize brushless dc permanent magnet (DC-PM) motors. Digital servo control by means of microprocessors is practical for many actuation systems. Systems of this type have a unique configuration and peculiar design requirements.

Near term technology (1990) will provide additional advances in EMAS, although the previous statements will remain unaltered. In anticipation of this, an advanced EMAS is presented in the following sections.

2. DESCRIPTION

A block diagram of a position servo system is shown in Figure B-1. The principal components are the controller, inverter, and actuator. Interfaces between the components are shown, also. The function of each component is briefly described below:

- Controller - The functions of the controller, generally, are (a) close the inner and outer servo response loops, (b) provide a motor current command to the inverter, and (c) provide an interface with the FCS.

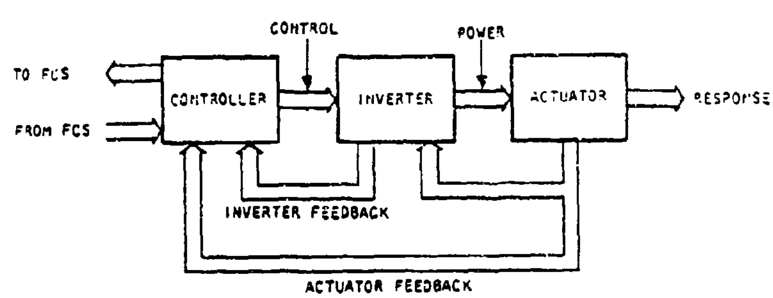


Figure B-1. Position Servo Block Diagram and Component Definition

Additional functions (fault detection, built-in-test, adaptive control) may be included in the controller for a particular application.

- Inverter - The functions of the inverter are to provide (a) motor commutation, (b) motor torque/speed control, (c) motor current limiting, and (d) motor drive electronics (power transistors, snubber circuitry, etc.) cooling. Again, additional functions may be included.

- Actuator - The actuator serves a single function, to convert electrical power to a mechanical response as a funtion of the servo command. The configuration of the actuator will be totally dependent upon the FCS requirements; but generally consists of motors and mechanical drives as a minimum. Each of the above are addressed in more detail in the following sections.

3. CONTROLLER

A more detailed block diagram of a general actuation system is shown in Figure B-2. This block diagram is representative of advanced (1990) technology.

The controller consists of blocks 1 through 6. Block 1 generates the speed command of the servo motor; block 2 generates the current command of the motor; block 3 allows the servo control laws to be altered during operation; and block 4 provides a monitor function for the FCS. Blocks 5 and 6 provide compensation for the rate and position feedback loops.

Each block may be digital or analog. However, to implement the above control scheme, digital appears to be the most viable. This is due to the availability of high speed microprocessors; and the large number of discrete components required by an equivalent analog system.

Interface with the FCS and EMAS components may be electrical or optical.

4. INVERTER

The inverter consists of blocks 7 through 10 in Figure B-2. Block 7 generates a pulse train as a function motor current error; block 8 contains commutation and current control switching logic, and power switch driver electronics; block 9 contains the power switches (transistors) required for motor commutation, torque/speed control, and current limiting; and block 10 senses motor current and provides feedback compensation in the current loop. Each of these functions will be explained more fully.

The mechanization of the above loops have been examined previously [5], and will remain analog for the far term. This is due to the frequency response requirements of the current limit loop. A digital system would require a microprocessor with a calculation rate in the gigahertz range. This is considered very unlikely during the near term.

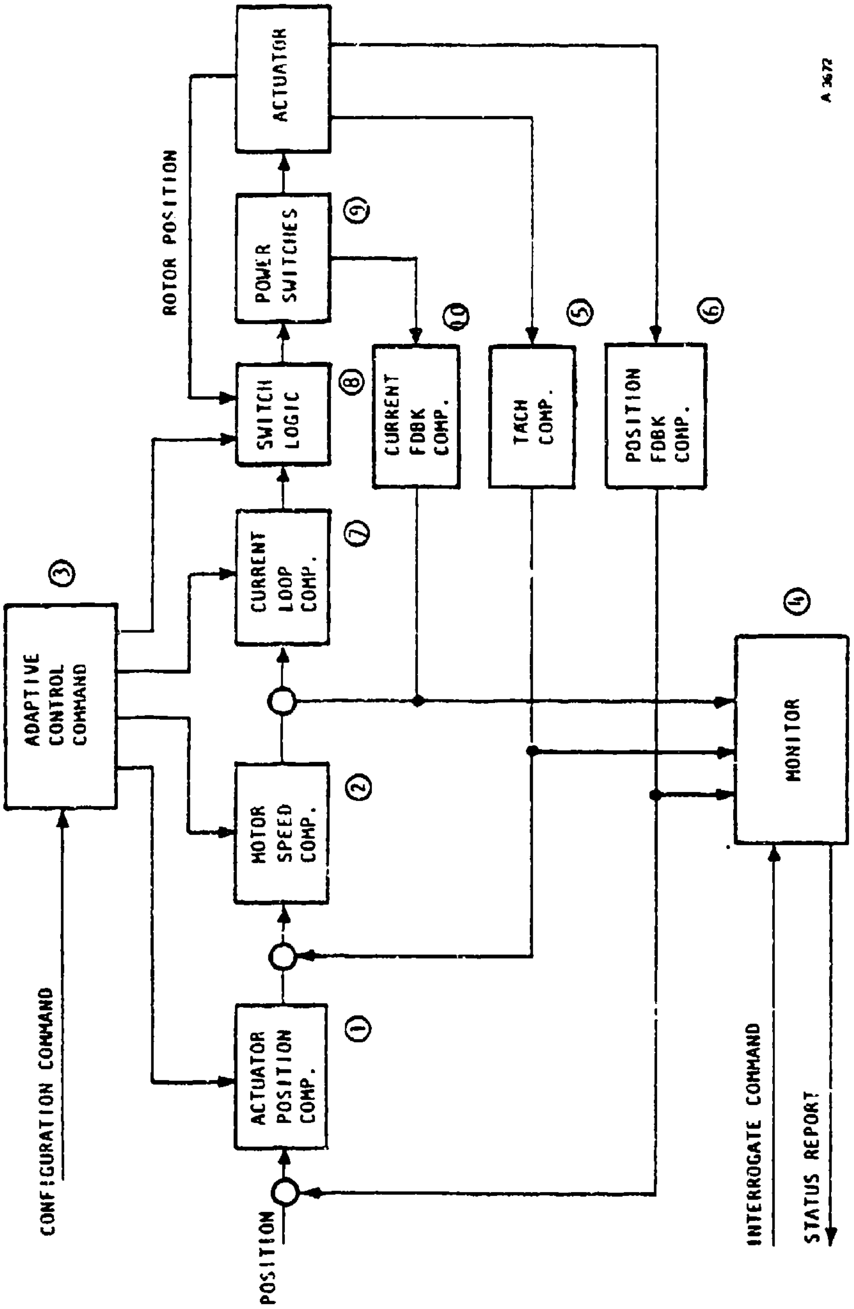


Figure B-2. Advanced Technology System Block Diagram

Interface may be accomplished by electrical or optical means; but electrics appears to be most desirable due to the analog control of the current loop.

4.1 DESCRIPTION

The principal function of the motor inverter assembly is to electrically commutate and control motor rotation over its entire speed range. A general inverter assembly block diagram is shown in Figure B-3.

The transistor inverter converts the 270 vdc source power to variable frequency 3-phase current applied directly to the motor windings. Figure B-4 shows the basic three phase motor drive waveforms. The sequence in which the transistors switch is conveyed by the switch number, S1 thru S6.

The inverter assembly may also encompass an input filter for conducted EMI suppression and energy storage; current sensing for feedback to the internal current control; drive logic for transistor control; motor position sensing for feedback to the switch logic; and although not shown, logic low level power supplies for control and protection functions.

4.2 OPERATION

The conversion of dc power into ac power is accomplished by six switch circuits as indicated in Figure B-4. Each switch (S1 thru S6) consists of a transistor and voltage limiting or snubber circuits (R1, C1, and CR1) as shown in Figure B-5. Internal to the transistor is a parallel free wheel diode that provides a path for the motor lagging reactive current flow. Each transistor switch can conduct up to a 120 electrical degree interval. During this interval the transistors are modulated (on-to-off) to control the current flow to the motor, thus designated transistors operate in a chopping mode to affectively pulse-width-modulate the ac output voltage amplitude. This technique inserts notches into the output waveform that cause a reduction in the fundamental current amplitude without causing an objectionable increase in the higher order harmonics.

The inverter is controlled such that the ac current is synchronized with the rotation or phase via motor rotor position sensors. The control logic uses motor position sensor information to force synchronization between the inverter current and motor phase rotation. In this manner, the correct relationship between motor generated EMF (electromotive force) and applied current from the inverter is continuously maintained.

Figure B-6, Figure B-7, and Figure B-8 illustrate the switching and waveforms involved. Figure B-6 shows the two types of switching, conducting and chopping. Conducting switching is used for basic commutation; chopping is used for current control. Figure B-7 shows the different voltage waveforms and resulting current waveform which exist in the inverter and motor. Figure 8 combines the result of these waveforms over 360 electrical degrees of machine rotation.

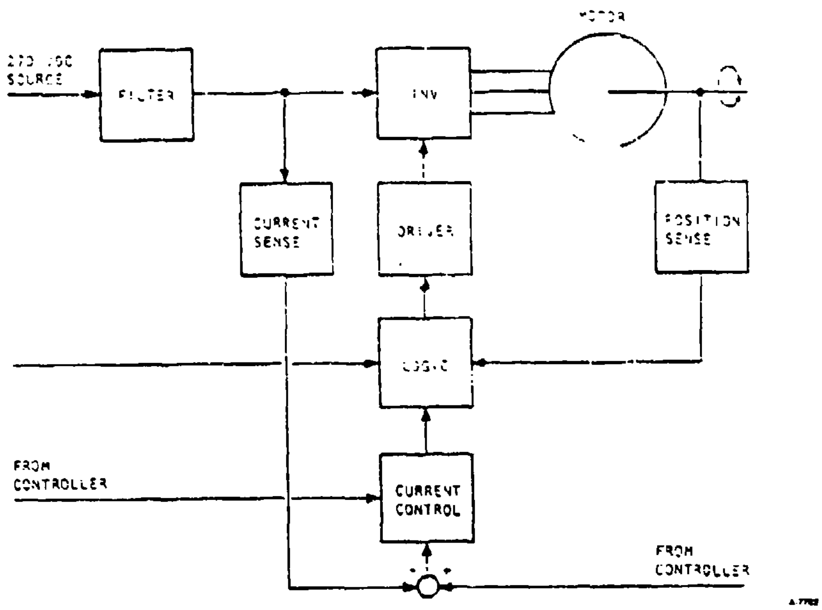


Figure B-3. Inverter Assembly Block Diagram

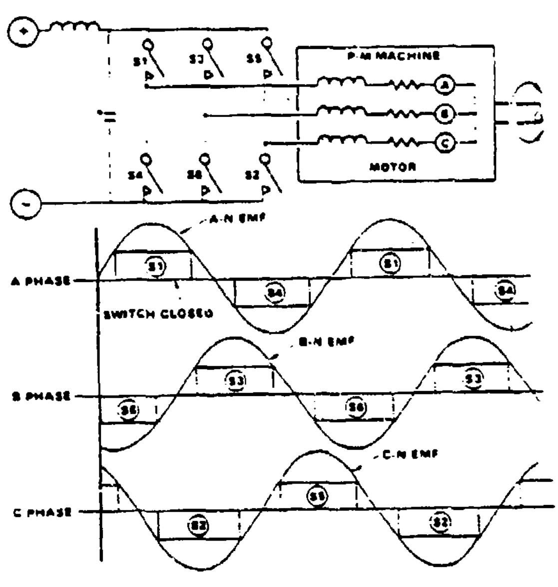


Figure B-4. Basic Motor Drive Waveform

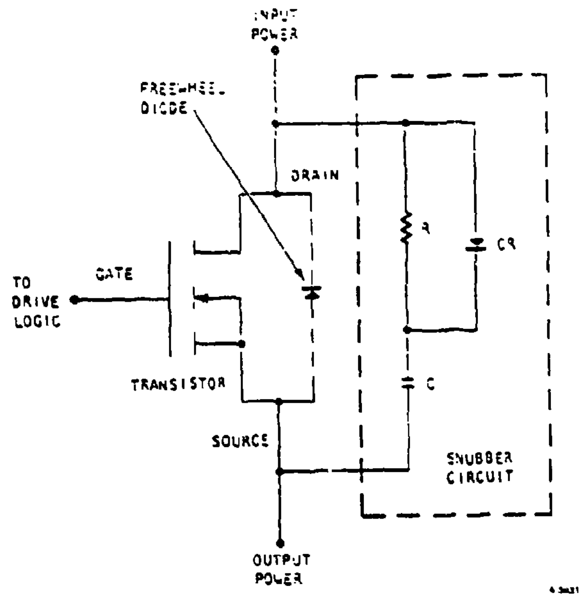


Figure B-5. Inverter Switch Schematic

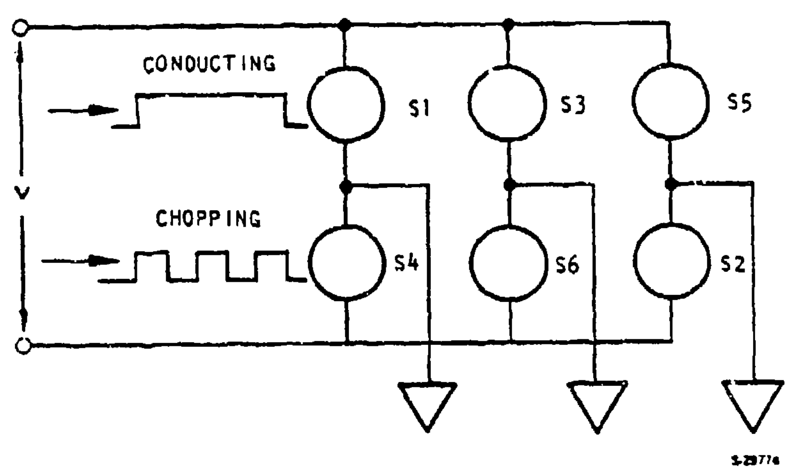


Figure B-6. Inverter Switching

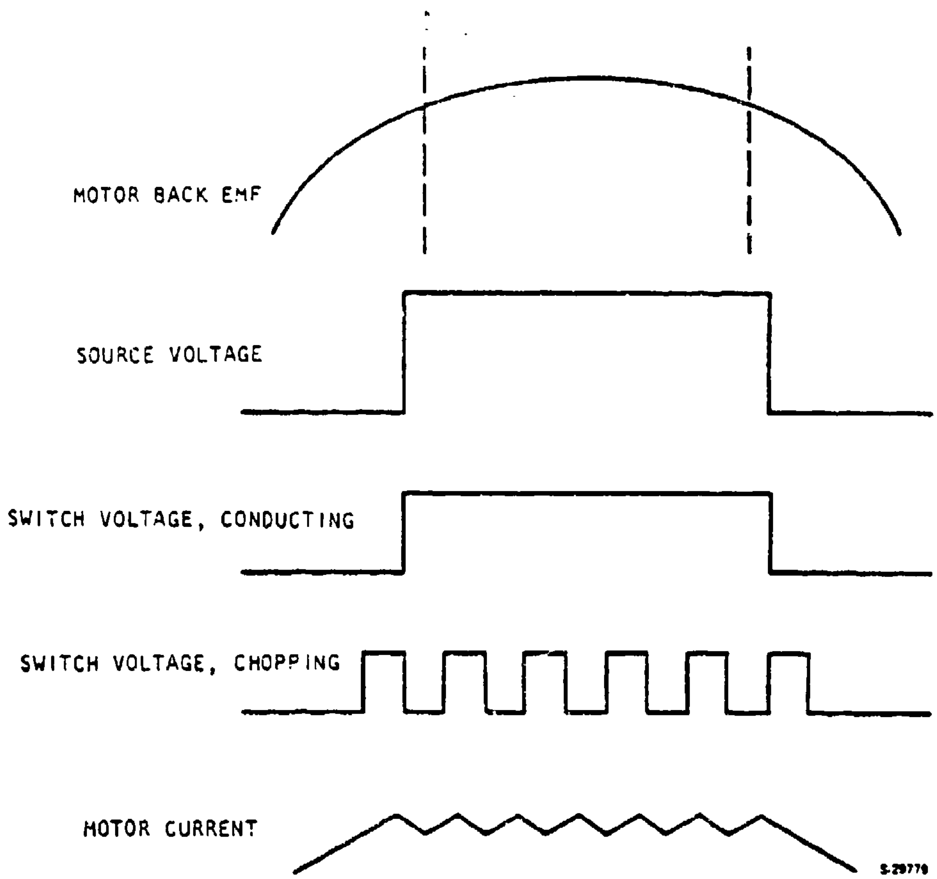


Figure B-7. Inverter/Motor Waveform

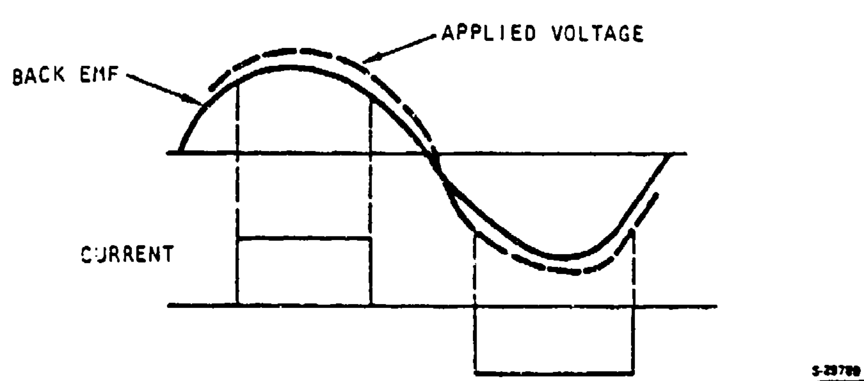


Figure B-8. Back EMF and Current Relationship

The chopped waveform is the result of a pulse-width modulation circuit (fixed frequency, variable width). This voltage waveform allows the control of current to a near constant value, as shown in Figure B-7 and Figure B-8.

Each of the switching waveforms (conducting and chopping) may be restricted to particular sets of switches (as implied in Figure B-6); or rotated among all the switches, for improved thermal balance.

4.3 THERMAL MANAGEMENT

Power inverters require cooling due to losses developed during transistor switching and conduction. Figure B-9 illustrates three types of cooling applicable to solid state inverters.

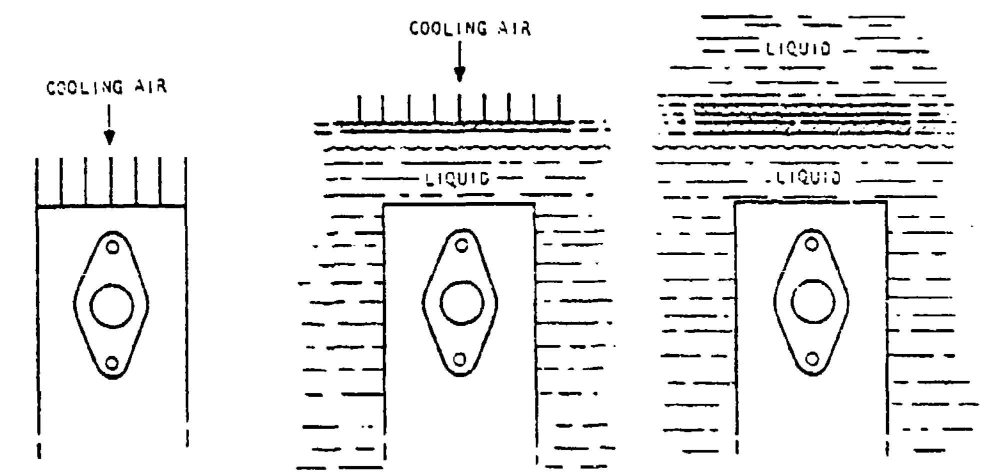


Figure B-9. Inverter Cooling Techniques

Evaporative cooling of electronic devices has grown increasingly popular in recent years as current density ratings of devices have increased [5]. In comparison with conventional air-cooling, the advantages that evaporative cooling offers are significant. They include:

(a) Fewer electronic components - The improved cooling increases the load-handling capability of a device, hence reducing the number of the devices required.

(b) Decreased weight and size - Evaporative cooling reduces the cooling system size and weight which normally forms the bulk of the electronic package.

(c) Higher reliability - Electronic devices are immersed in a dielectric liquid and are not susceptible to dust fouling and resultant problems. The cooling system is inherently more reliable when there is no forced-air involved.

(d) Less maintenance - Periodic maintenance operations, such as dust cleaning, are not required for electronic components and heat sinks.

(e) Less noise - Noise levels will be very low or non-existent.

4.4 POWER TRANSISTORS

Power transistors have traditionally been the principal limiting factor in the application of brushless DC-PM motors, where significant current levels were required. Transistors are currently available which have very high current ratings; but do not have all the desirable characteristics a designer would like.

Power FET's (field effect transistors) offer potential for the required high in current ratings, in the near future [7]. Existing desirable features of power FET's include short switching times, low switching losses, and excellent load sharing.

T-I summarizes probable characteristics of the near term device.

T-I Near Term FET Characteristics

FET current rating	50 amp
FET on-state resistance (R_{DC}) at maximum junction temperature of 150°C	0.10 ohm
FET diode forward voltage drop (V_{CR}) at 50 Adc	1.5 vdc
FET current rise (t_r) and fall (t_f) time for 50 amp-dc changes	50 nsec
FET thermal resistance from junction to case	0.42°C/watt
FET gate to source capacitance	1000 pF
Lead inductance from the FET assembly to a voltage source	0.3 H
Maximum allowable FET drain to source voltage caused by switching	320 volts

5. SUMMARY

A general description of a near term EMAS has been presented. Construction, operation, and technology assumptions have been stated.

It is emphasized that there are numerous methods for mechanizing each component of the EMAS (controller, inverter, actuator). Individual applications may dictate specialized approaches to system control. For FCS applications in the near term however, the presented approach appears to be attractive and viable.

REFERENCES

1. Wood, Ecnolds, Ashmore - AFFDL Report No. TR-75-42 Electromechanical Actuation Feasibility Study.

2. Grau - MACC Report No. 75170-30 Feasibility Investigation for Advanced Flight Control Actuation Systems: All Electric Concepts (AF CAS-AE).

3. Wood, Lewis - AFFDL Report No. TR-73-150 Electromechanical Actuation Development.

4. Lewis, Gray, Wood - AFWAL Report No. TR-80-3024 Electromechanical Actuation Development.

5. Rowe - AiResearch Report No. 80-17284 Electromechanical Airplane Actuation Trade Study.

6. Rollet - Technical Paper "Two Phase Freon Cooling for Electronic Power Equipment".

7. Blanchard, Haynie - Power Conversion International, V.6, No. 2, Mar/Apr 1980 "Power MOS Transistors: Structure and Performance.

APPENDIX C

AIRPLANE ACTUATION TRADE STUDY

This appendix includes the work done by AiResearch, under the direction of S. Rowe, as part of their effort under service agreement L9FM-11231-405 to finalize the definition of the controllers and inverters needed for implementing the "all-electric" (Aircraft I) approach to the trade study. Work included in this appendix also further refines the weight and envelope data for the inboard flap actuator as well as those for the other major flight control actuation functions. These latter data were used directly in the preparation of Figures 25 through 31.

Appendix C

All Electric Airplane Study
Service Agreement L9FM-11231-405
Progress Report for the Period October 1980 - January 1981:
Advanced Electromechanical Actuation Systems

PREFACE

This document is submitted in conjunction with Service Agreement L9FM-11231-405, All Electric Airplane Study. The data submitted herein is presented to facilitate completion of the subject study. This document, in addition to previously submitted data, shall serve as a progress report for the period of October 1980 - January 1981.

INTRODUCTION

Several objectives of the statement-of-work have been completed during this reporting period. Specific accomplishments are:

- Identification of actuation system performance requirements
- Preliminary design of candidate actuators, and candidate selection
- Inverter analysis and design
- Controller concept selection and sizing

Each of the above are addressed in following sections.

REQUIREMENTS

Performance requirements for the actuation systems were defined by the customer during 1980 [3]. Subsequent discussions between AiResearch and the customer modified the flight control system (FCS) actuation system performance requirements to less stringent, but wholly satisfactory criteria.

A summary of FCS actuation system performance may be found in Appendix A.

ACTUATORS

Actuator performance and other design requirements were examined [1]. Using these data, preliminary designs for FCS candidate actuation systems were generated using the following approach:

- Motors were sized based on duty cycle, and steady-state and dynamic performance requirements
- Mechanical drives (output reduction) were sized based on peak load and life requirements

- Gearheads (intermediate reduction) were sized based on gear-ratio and load requirements

- Actuator dimensions and weights were calculated using the preceding data

Assumptions used to establish actuator configurations were:

- All motors are a brushless direct current-permanent magnet (DC-PM) configuration

- Motor magnets have an energy product (BH) of 22 X 10^6 Gauss-Oersted

- All reduction is simple planetary, compound planetary, or ballscrew

Candidate designs may be found in Appendix B. A final selection was made after consulting the customer. Selections are indicated in the appendix.

INVERTERS

Inverter sizing is usually determined by the type of cooling employed. Natural radiation and convection (heatsink) techniques are a first choice due to their simplicity and low cost. However, size and weight become unacceptable at higher power levels (5-6 kw inverter rating) and alternate cooling schemes must be investigated.

Inverter current requirements for each of the actuation system motors were determined, and may be found in Appendix C. The inverter configuration previously submitted was assumed, and is discussed in Reference [1]. Devices were selected based on current requirements, resulting in two power field effect transistors (FET) with ratings of 25 and 50 amps. FET characteristics and rationale are presented in Reference [1] (the 50 amp device represents 1990 state-of-the-art).

Cooling requirements were determined next, by calculating inverter losses as a function of duty cycle for each actuation system. Cooling techniques were evaluated, and the following conclusions reached:

- All systems requiring a 25 amp or smaller inverter may be cooled by natural radiation and convection

- All other systems requiring more than 25 amps required alternate techniques, and evaporative cooling was selected

Evaporative cooled inverters suitable for the required application had been previously sized in a similar study [2]. Inverter configuration is depicted in Figure C-1. Existing inverter dimensions were scaled to take into account applicable boundry conditions, and are tabulated in Table C-1. Note that forced convection with surface finning was assumed for inverter cooling.

It may be possible to eliminate any external cooling requirement by increasing inverter fin area and coolant mass. This will require as a minimum, simulation of inverter transient thermal response.

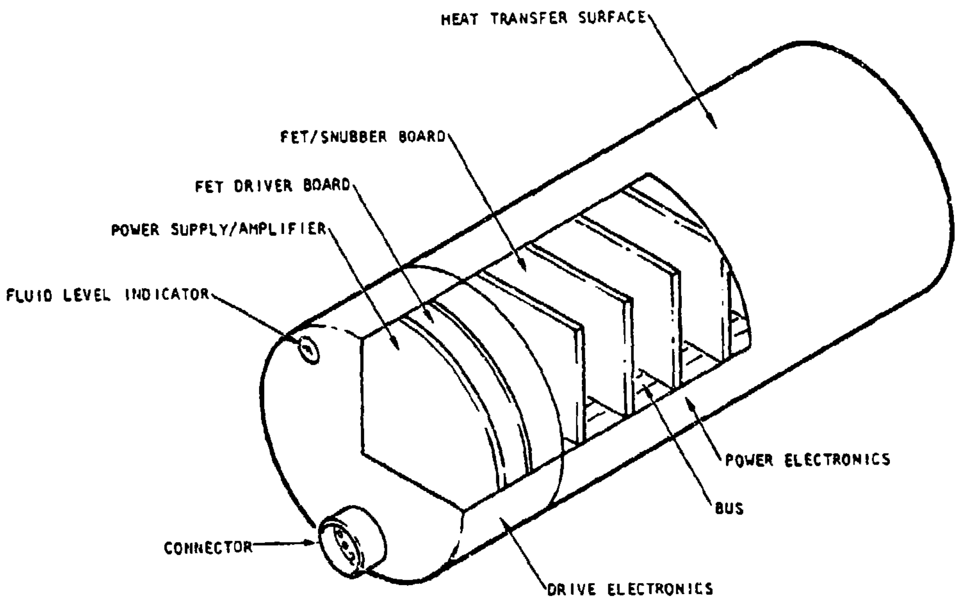


Figure C-1. Evaporative Cooled Inverter (No Fins)

Table C-1

Evaporative Cooled Inverter Data*

Actuation System	Current Rating	Diameter	Length	Weight
Inboard Flap	150 amp	7.2 in	18.7 in	38.0 lb
Midspan Flap	50	5.0	10.4	12.5

*Finned surface with forced convection, 130°F, 50 cfm, 2 in-H_2O P

Subsequently, the remaining inverters were sized using individual heatsinks for FET cooling. Figure C-2 shows the heatsink used, and Figure C-3 illustrates inverter configuration. A common design was used for all remaining actuation systems in light of the small size of the 25 amp inverter. Table C-2 summarizes inverter characteristics.

A summary of the thermal analysis for the each inverter configuration may be found in Append'x D to this appendix.

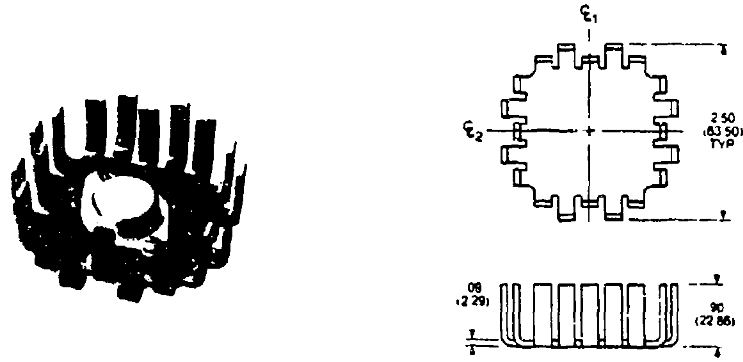


Figure C-2. Heatsink Configuration

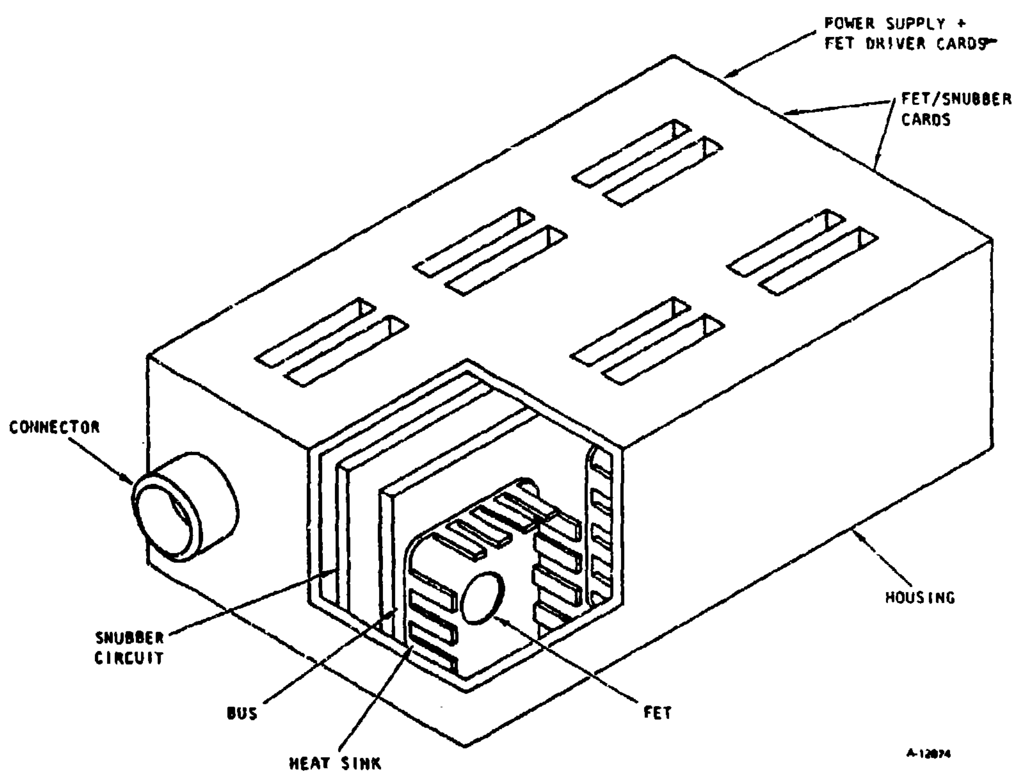


Figure C-3. Natural Radiation/Convection Cooled Inverter (Typical)

Table C-2

Natural Radiation/Convection Cooled Inverter

Actuation System	Current Rating	Length	Width	Depth	Weight
Aileron, Canard, Leading Edge Flap, Upper/Lower Rudder	25 amp	11 in	7 in	3 in	10 lb

CONTROLLER

A controller was configured for multiple actuator control, using the concept of Reference [1]. Up to six inverters and motors, and four actuators may be interfaced with the unit. This approach allowed the use of a common controller for all 270 vdc servos. Controller characterics are summarized in Table C-3.

Figure C-4 is a block diagram of the controller. The unit is full DMX (digital multiplexed). All interfaces are shown as optical, although if transmission distances are short, electrical interfaces may be desirable. A feasible scheme would be optical for the FCS, motor, and actuator interfaces; and an electrical interface for the inverters.

Table C-3

CONTROLLER SUMMARY

Dimensions	4 x 4 x 8 in.
Weight	5 lb
Cooling	Air
Interface	Optical/Electrical Bus, Power
Function	Supports up to 6 inverters, 6 motors, and 4 actuators

Two way buses are used for the FCS and inverter interface only, all other buses are one way. This was chosen since only feedback data is necessary from the motors and actuators.

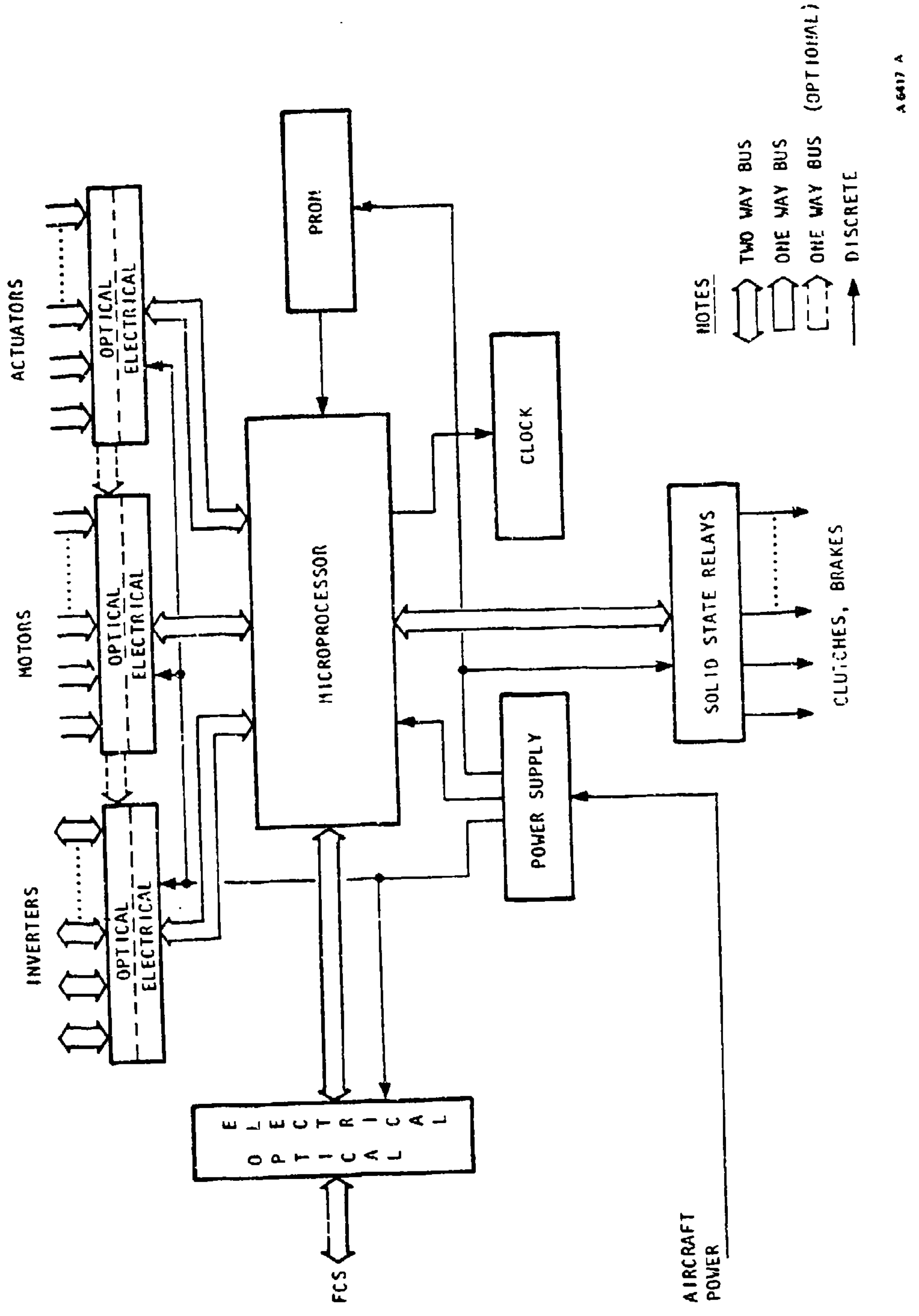


Figure C-4. Controller Block Diagram

Controller interface is shown in Figure C-5. Note that all motor and actuator data flow to the controller. Motor data is required by the inverter for commutation (rotor position), so any necessary data are passed on to the inverter by an optical-optical or optical-electrical coupling, as required. Figure C-5 indicates that any actuation system data may be made directly available to the inverter, as necessary. Table C-4 summarizes FCS-controller interface.

Current limit control and commutation logic were assumed to be integral with the inverter. A multiplexed interface (optical or electrical) would also be required at the inverter.

Control of the clutches and brakes of the various actuation systems was assumed to be performed at the controller. Discrete wiring was also assumed for clutch/brake power. The devices are controlled by solid state relays located in (or near) the controller. Figures C-4 and C-5 illustrate interface and operation.

PROJECTED PROGRESS

During the next reporting period a detailed design of one FCS actuator will be performed. The customer will be consulted prior to the start of the detailed design, for actuation system selection and envelope specification

REFERENCES

1. Rowe--AiResearch Report No. 80-17351(1), All Electric Airplane Study, Progress Report for the Month of September

2. Rowe--AiResearch Report No. 80-17284, Electromechanical Airplane Actuation Trade Study

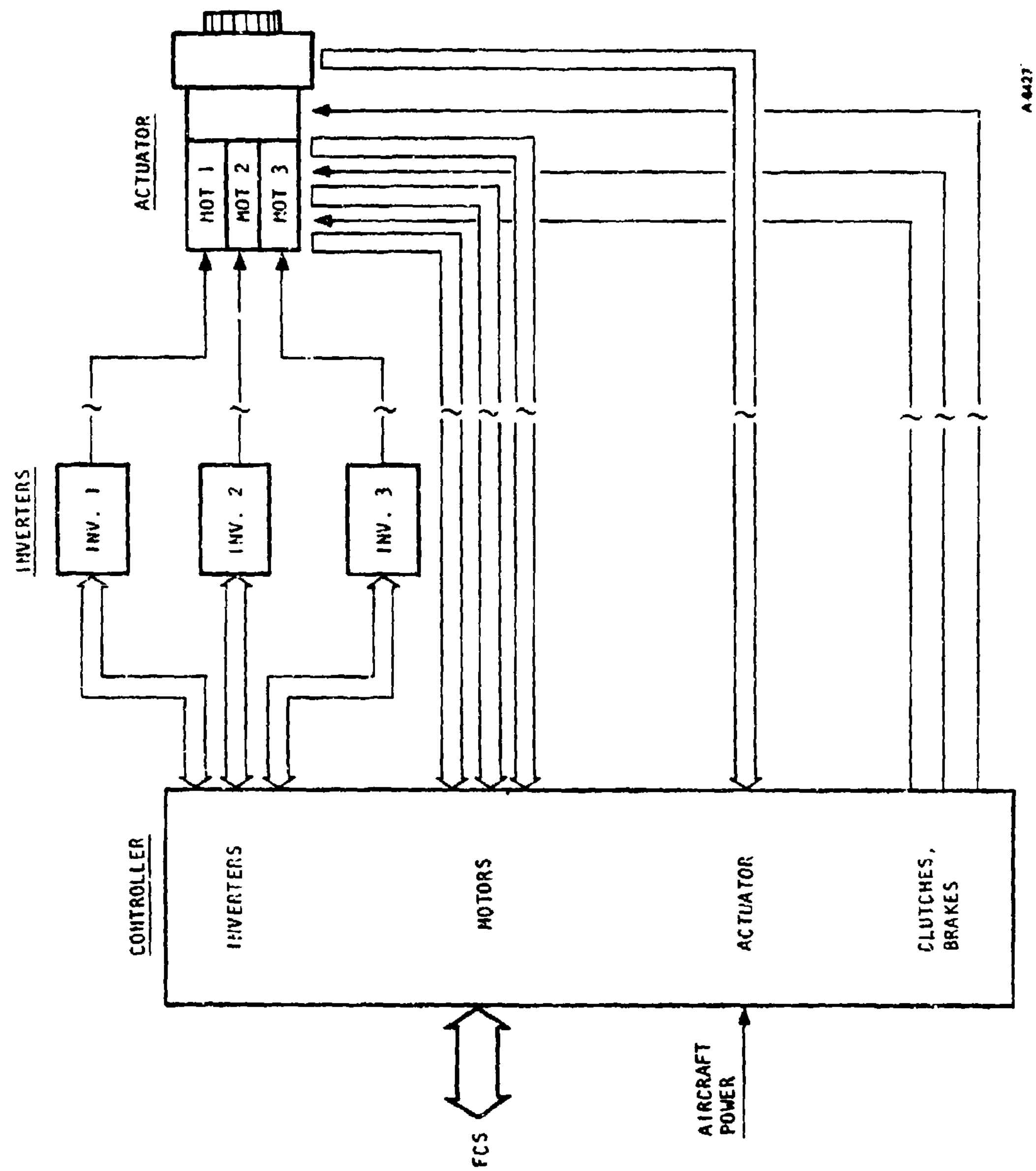


Figure C-5. Controller Interface (Typical)

Table C-4

Controller Interface Requirements

Aircraft-Controller

- 1-2 way or 2-1 way data bus
- 270 vdc

Controller-Inverter

- 1-2 way or 2-1 way data bus
- 270 vdc

Inverter-Motor

- 3-phase wiring

Controller-Motor

- 1-way data bus*

Controller-Actuator

- 1-way data bus
- Discrete wiring (1 per clutch/brake)
- 270 vdc

*May be possible to reduce to 1 bus per actuator

Appendix A

Actuation System Requirements

DATE 9-11-80
PART NO. RAATS
PREPARED BY S. ROWE

Appendix C

CALC. NO. 1-001 SHEET NO. 1
MODEL NO.
CHECKED BY

ACTUATOR REQUIREMENTS*

	ACTUATOR	TRAVEL	LOAD CHARACTERISTICS				REDUN-DANCY PER SURFACE	ACTR'S -OR- CHANNELS PER SURFACE
			NO-LOAD RATE	DESIGN LOAD & RATE	STALL LOAD	DESIGN POWER		
(1)	INBOARD FLAP	-30° +45°	100 DPS	113 KIP 50 DPS	151 KIP	14.94 HP 11.15 KW	FO^2	3
(2)	MIDSPAN FLAP	-30° +45°	100 DPS	23 KIP 50 DPS	30 KIP	3.09 HP 2.27 KW	FO^2	3
(3)	AILERON	±25°	50 DPS	21 KIP 20 DPS	26 KIP	1.11 HP 0.83 KW	FO	2
(4)	UPPER RUDDER	±20°	40 DPS	7 KIP 30 DPS	9 KIP	0.56 HP 0.41 KW	FO^2	3
(5)	LOWER RUDDER	±20°	40 DPS	10 KIP 20 DPS	13 KIP	0.53 HP 0.39 KW	FO^2	3
(6)	L.E. FLAP	-0° +20°	5 DPS	153 KIP 2.4 DPS	173 KIP	0.97 HP 0.72 KW	FO	2
(7)	CANARD	15.36 IN	2.3 IPS	2.0 KLB 0.77 IPS	2.4 KLB	0.23 HP 0.17 KW	FO	2

*BASED ON MINUTES FROM MEETING AT ROCKWELL ON 8-28-80

Appendix B

Actuation Preliminary Designs

DATE 9-22-80	Appendix C	CALC. NO. 2-001	SHEET NO. 1
PART NO. RAATS		MODEL NO.	
PREPARED BY S. ROWE		CHECKED BY	

INBOARD FLAP INSTALLATION

USE THE FOLLOWING DATA FOR INSTALLATION PURPOSES :

1. MOTOR

- DIAMETER 3.5 IN
- LENGTH 10.6 IN
- WEIGHT 12.0 LB

2. DRIVE

- DIAMETER 3.5 IN
- LENGTH 9.4 IN
- WEIGHT 11.1 LB

3. GEARHEAD

- DIAMETER ~ 5.5 IN
- LENGTH ~ 5.0 IN
- WEIGHT ~15.0 LB

} INVESTIGATE REDUCTION OF DIAMETER

4. CLUTCH

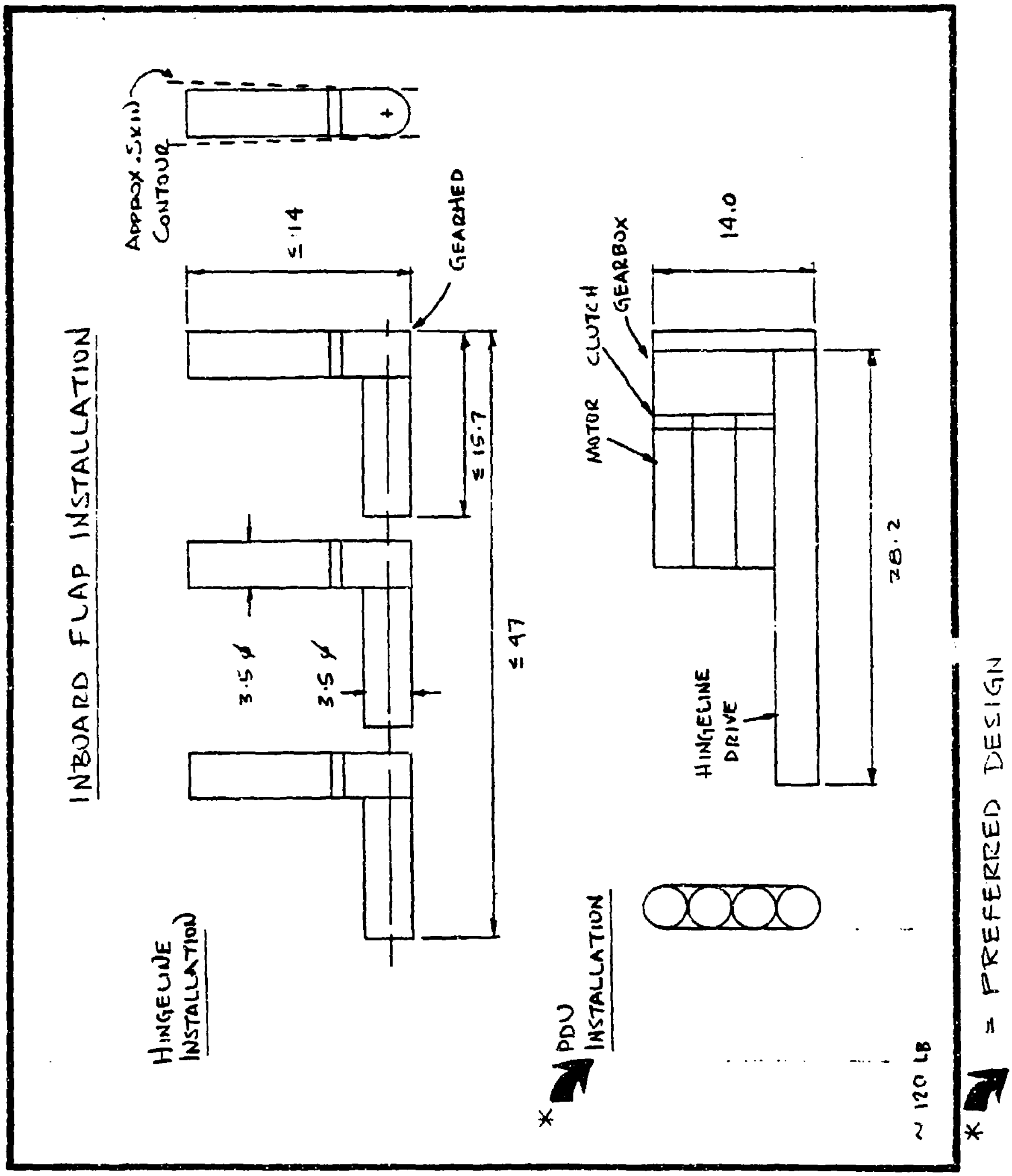
- DIAMETER ~3.5 IN
- LENGTH ~ 1.0 IN
- WEIGHT ~2.0 LB

SEE ATTACHED FOR INSTALLATION.

DATE 9-22-80 | Appendix C | CALC. NO. 2-001 SHEET NO. 2

PART NO. RAATS | MODEL NO.

PREPARED BY S. ROWE | CHECKED BY



DATE 9-22-80 | Appendix C | CALC. NO. Z-002 SHEET NO. 1

PART NO. RAATS | MODEL NO.

PREPARED BY S. ROWE | CHECKED BY

MIDSPAN FLAP INSTALLATION

USE THE FOLLOWING DATA FOR INSTALLATION PURPOSES:

1. MOTOR

• DIAMETER	2.3 IN
• LENGTH	7.7 IN
• WEIGHT	2.3 LB

2. DRIVE

• DIAMETER	1.6 IN
• LENGTH	9.1 IN
• WEIGHT	2.9 LB

3. GEARHEAD

• DIAMETER	3.0 IN
• LENGTH	3.5 IN
• WEIGHT	6.0 LB

4. CLUTCH

• DIAMETER	~ 2.3 IN
• LENGTH	~ 1.0 IN
• WEIGHT	~ 1.0 LB

DATE 9-22-80

PART NO. RAATS

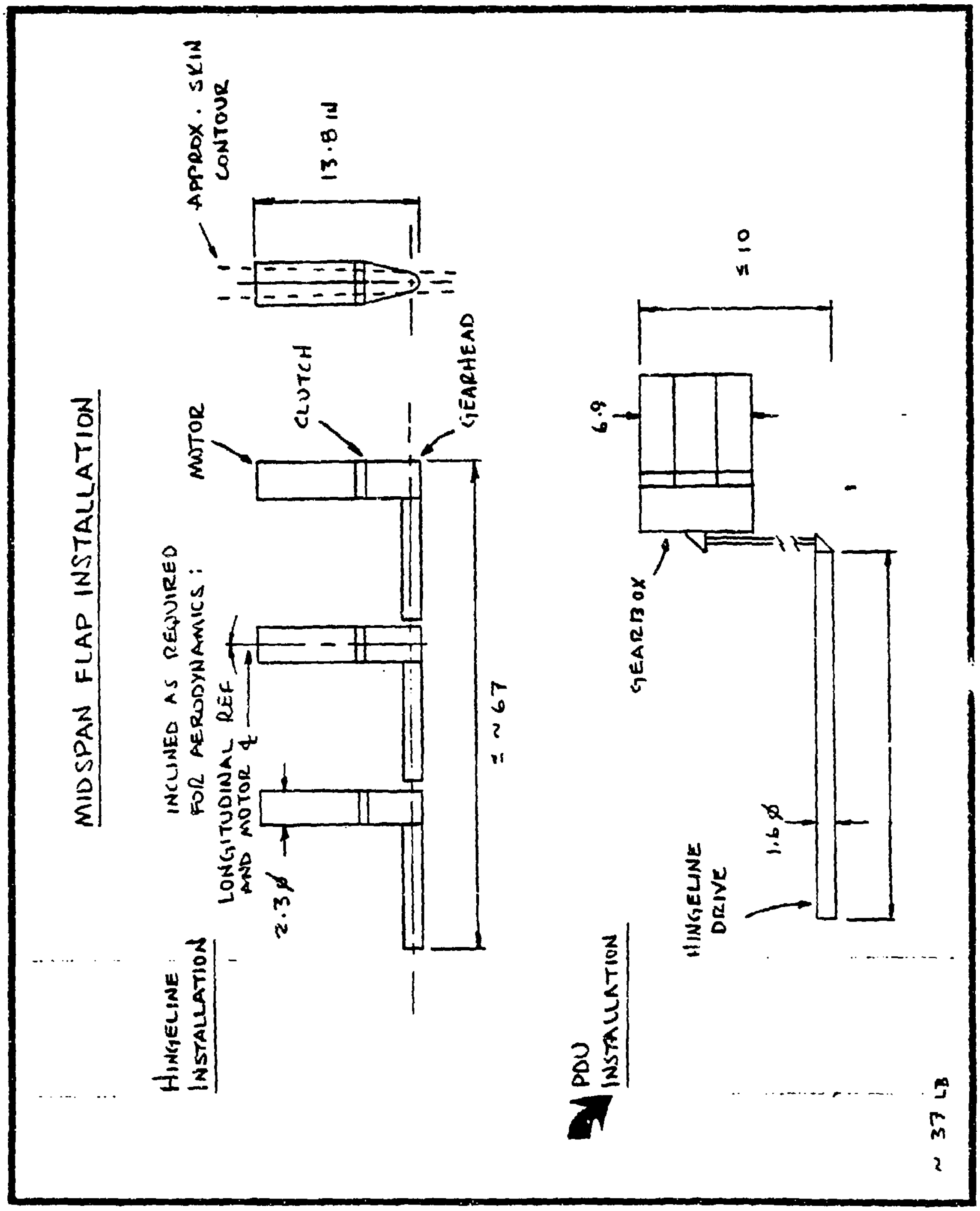
PREPARED BY S. ROWE

Appendix C

CALC. NO. 2-002 SHEET NO. 2

MODEL NO.

CHECKED BY



DATE 9-22-90	Appendix C	CALC. NO. 2-003	SHEET NO. 1
PART NO. RAATS		MODEL NO.	
PREPARED BY S. ROWE		CHECKED BY	

AILERON INSTALLATION

USE THE FOLLOWING DATA FOR INSTALLATION PURPOSES:

1. MOTOR
   - DIAMETER 1.9 IN
   - LENGTH 6.7 IN
   - WEIGHT 1.0 LB

2. DRIVE
   - DIAMETER 1.4 IN
   - LENGTH 11.8 IN
   - WEIGHT 3.2 LB

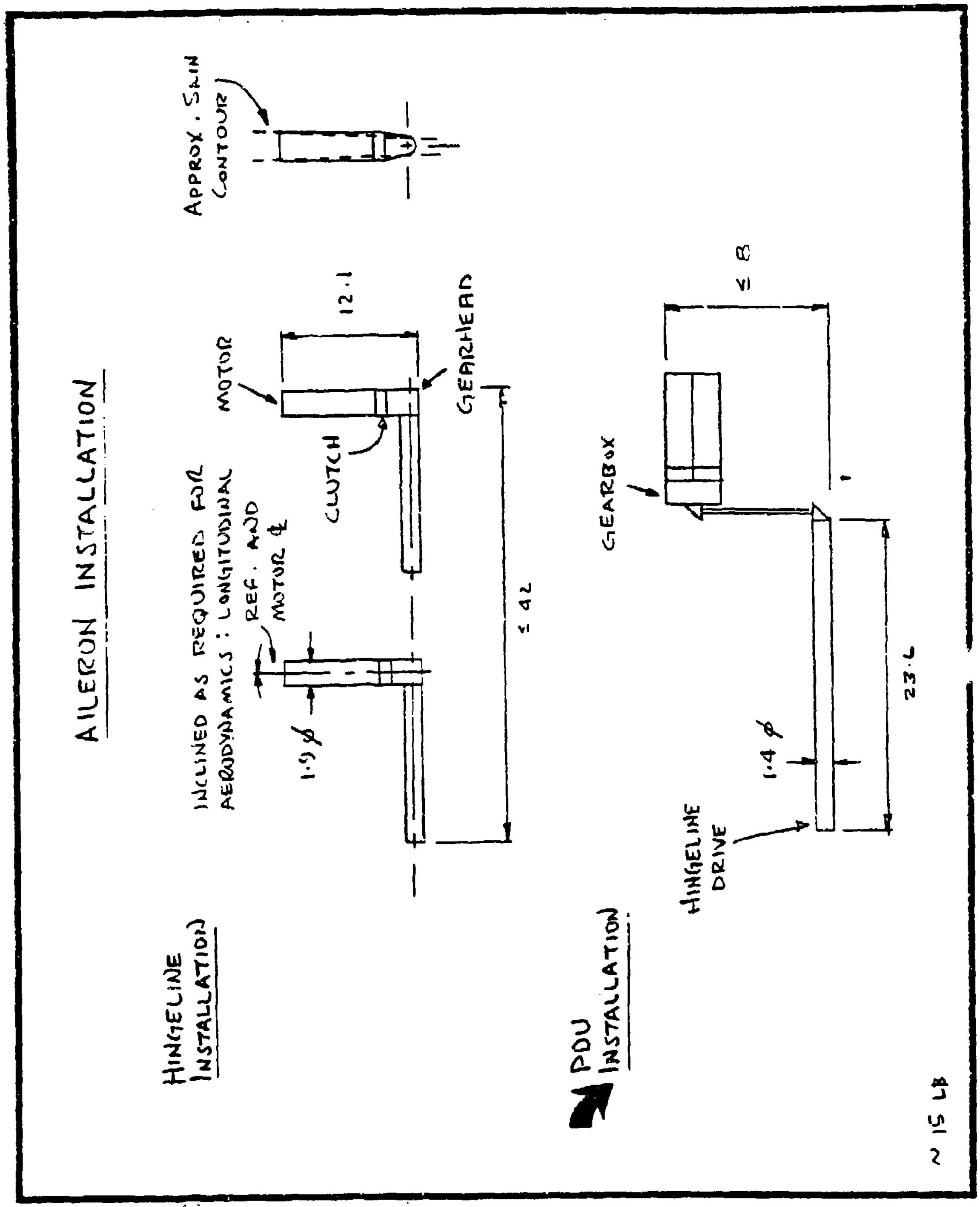
3. GEARHEAD
   - DIAMETER 2.5 IN
   - LENGTH 3.0 IN
   - WEIGHT 2.5 LB

4. CLUTCH
   - DIAMETER ~1.9 IN
   - LENGTH ~1.0 IN
   - WEIGHT ~0.8 LB

DATE 9-22-80 | Appendix C | CALC. NO. 2-003 SHEET NO. 2

PART NO. RAATS | MODEL NO. ______

PREPARED BY S. ROWE | CHECKED BY ______



DATE 9-22-80 Appendix C CALC. NO. 3-004 SHEET NO. 1

PART NO. RAATS MODEL NO.

PREPARED BY S. ROWE CHECKED BY

UPPER RUDDER INSTALLATION

USE THE FOLLOWING DATA FOR INSTALLATION PURPOSES:

1. MOTOR

• DIAMETER	1.9 IN
• LENGTH	5.1 IN
• WEIGHT	0.3 LB

2. DRIVE

• DIAMETER	1.4 IN
• LENGTH	4.1 IN
• WEIGHT	1.1 LB

3. GEARHEAD

• DIAMETER	1.5 IN
• LENGTH	1.8 IN
• WEIGHT	1.0 LB

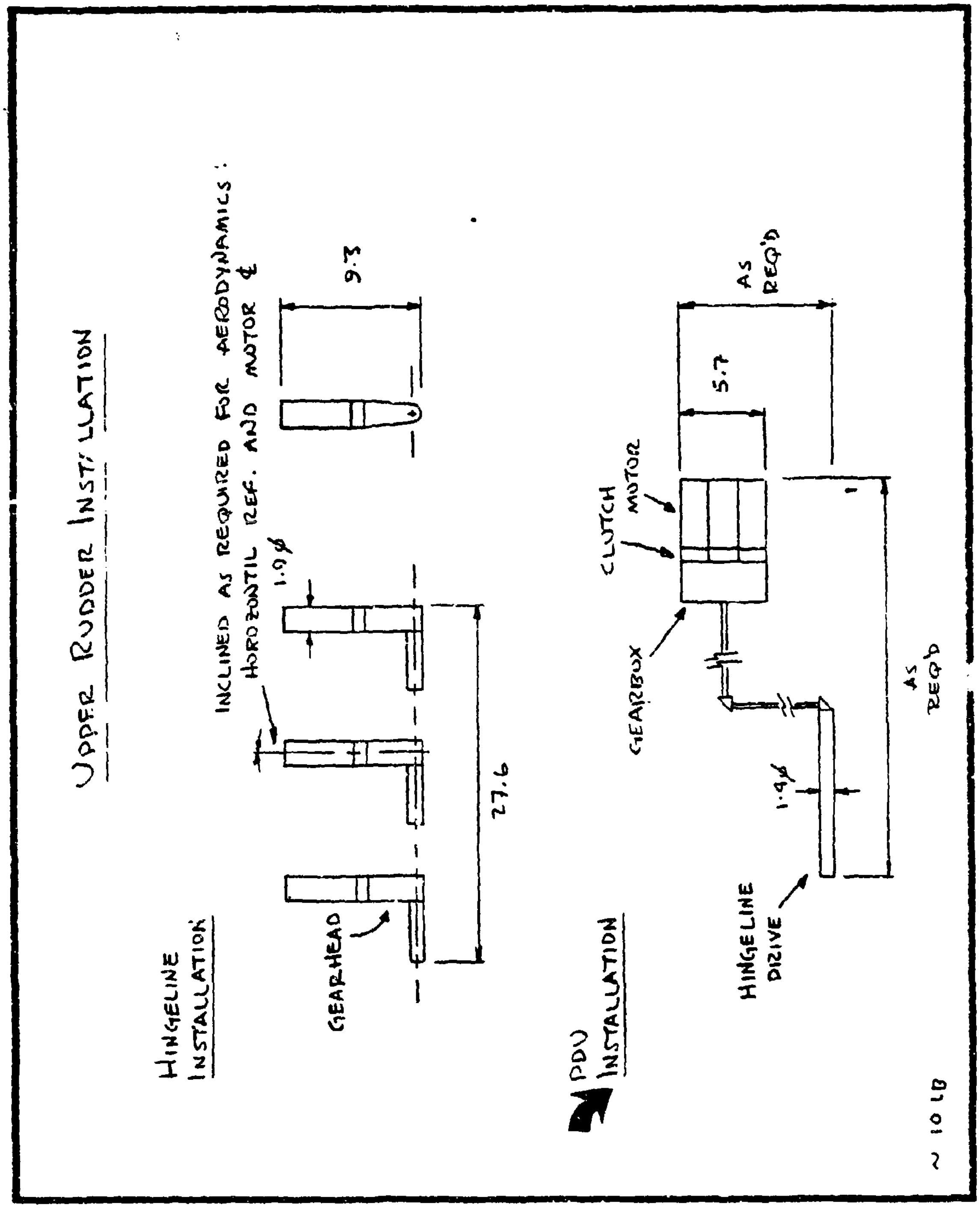
4. CLUTCH

• DIAMETER	~ 1.9 IN
• LENGTH	~ 1.0 IN
• WEIGHT	~ 0.8 LB

DATE 9-22-82 Appendix C CALC. NO. 2-004 SHEET NO. 2

PART NO. RAATS MODEL NO.

PREPARED BY S. ROWE CHECKED BY



DATE 9-22-80 | CALC. NO. 2-005 | SHEET NO. 1

PART NO. RAATS | Appendix C | MODEL NO.

PREPARED BY S. ROWE | CHECKED BY

LOWER RUDDER INSTALLATION

USE THE FOLLOWING DATA FOR INSTALLATION PURPOSES :

1. MOTOR

• DIAMETER	2.3	IN
• LENGTH	6.0	IN
• WEIGHT	2.6	LB

2. DRIVE

• DIAMETER	1.6	IN
• LENGTH	4.0	IN
• WEIGHT	1.3	LB

3. GEARHEAD

• DIAMETER	2.0	IN
• LENGTH	2.0	IN
• WEIGHT	2.0	LB

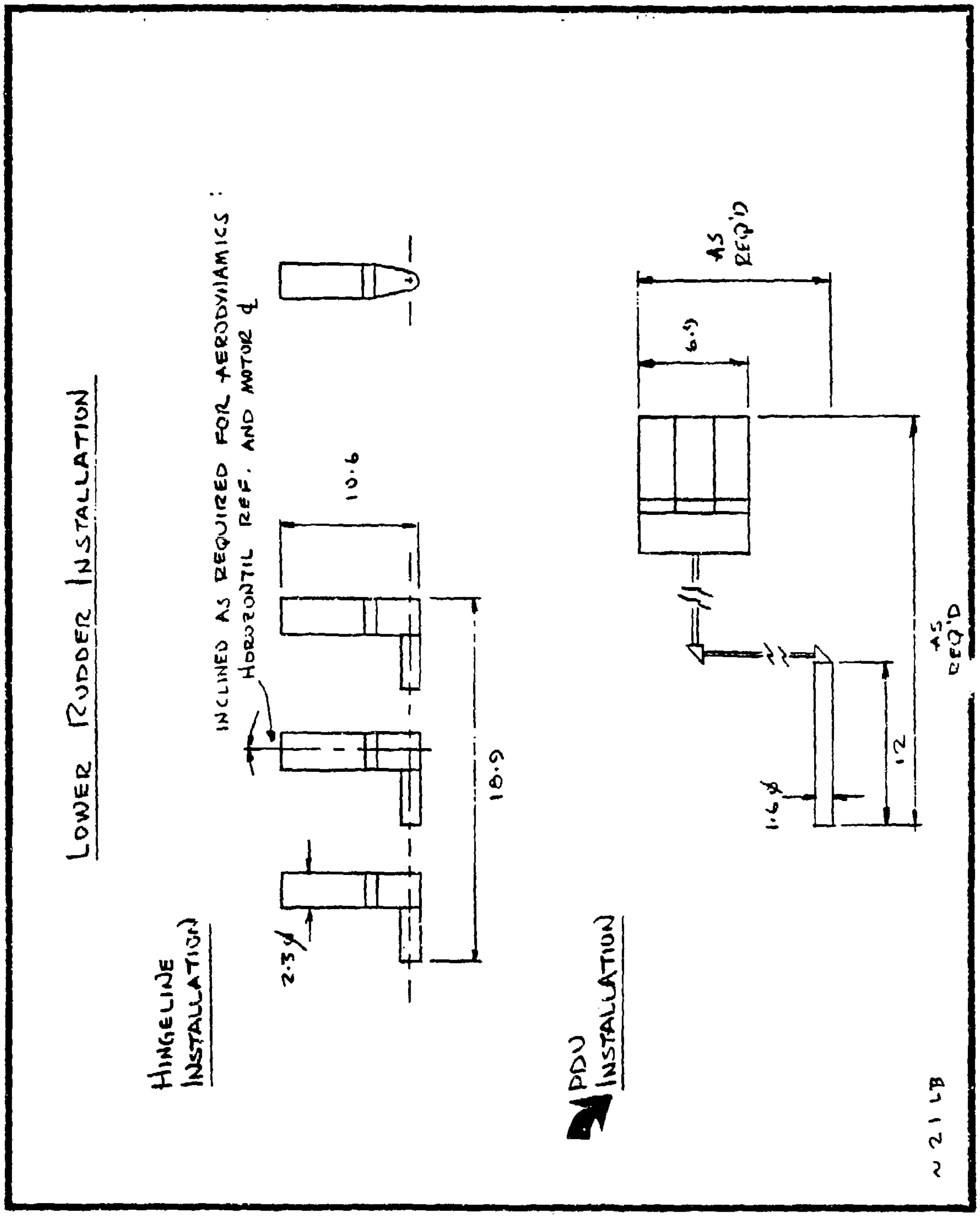
4. CLUTCH

• DIAMETER	~ 2.3	IN
• LENGTH	~ 1.0	IN
• WEIGHT	~ 1.2	LB

DATE 9-22-80 Appendix C CALC. NO. 2-005 SHEET NO. 2

PART NO. RAATS MODEL NO.

PREPARED BY S. ROWE CHECKED BY



DATE 9-22-80 | Appendix C | CALC. NO. 2-006 SHEET NO.

PART NO. TRAATS | MODEL NO.

PREPARED BY S. ROWE | CHECKED BY

L.E. FLAP INSTALLATION

USE THE FOLLOWING DATA FOR INSTALLATION PURPOSES:

1. MOTOR
   - DIAMETER 2.3 IN
   - LENGTH 8.1 IN
   - WEIGHT 2.6 LB

2. DRIVE
   - DIAMETER 4.0 IN
   - LENGTH 7.2 IN
   - WEIGHT 12.8 LB

3. GEARHEAD
   - DIAMETER 2.5 IN
   - LENGTH 3.0 IN
   - WEIGHT 3.0 LB

4. CLUTCH
   - DIAMETER ~ 2.3 IN
   - LENGTH ~ 1.0 IN
   - WEIGHT ~ 1.2 LB

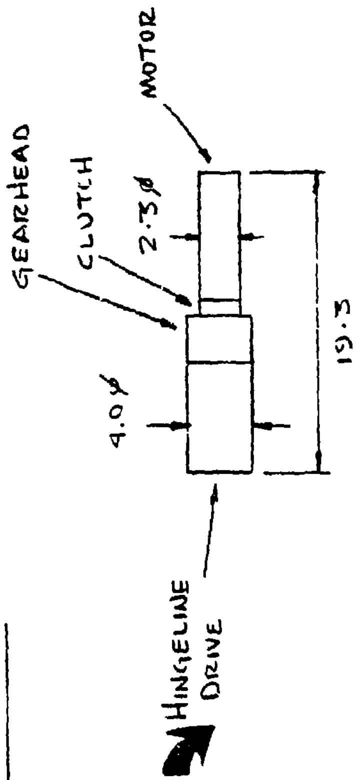
DATE 9-22-80 | Appendix C | CALC. NO. 2-006 SHEET NO. 2

PART NO. RAATS | MODEL NO.

PREPARED BY S ROWE | CHECKED BY

L.E. FLAP INSTALLATION

HINGELINE INSTALLATION



~20 LB (~240 LB FOR 12 ACTR'S)

DATE 9-22-80 CALC. NO. 2-007 SHEET NO. 1

PART NO. RAATS MODEL NO.

PREPARED BY S. ROWE CHECKED BY

CANARD INSTALLATION

USE THE FOLLOWING DATA FOR INSTALLATION PURPOSES:

1. MOTOR
   - DIAMETER 2.3 IN
   - LENGTH 5.3 IN
   - WEIGHT 0.5 LB

2. DRIVE
   - BALL ASSY OD 1.0 IN
   - SCREW OD 2.0 IN
   - TOTAL LENGTH 19.9 IN
   - WEIGHT 11.6 LB

3. GEARING
   - DIAMETER 1.5 IN
   - LENGTH 1.8 IN
   - WEIGHT 1.0 LB

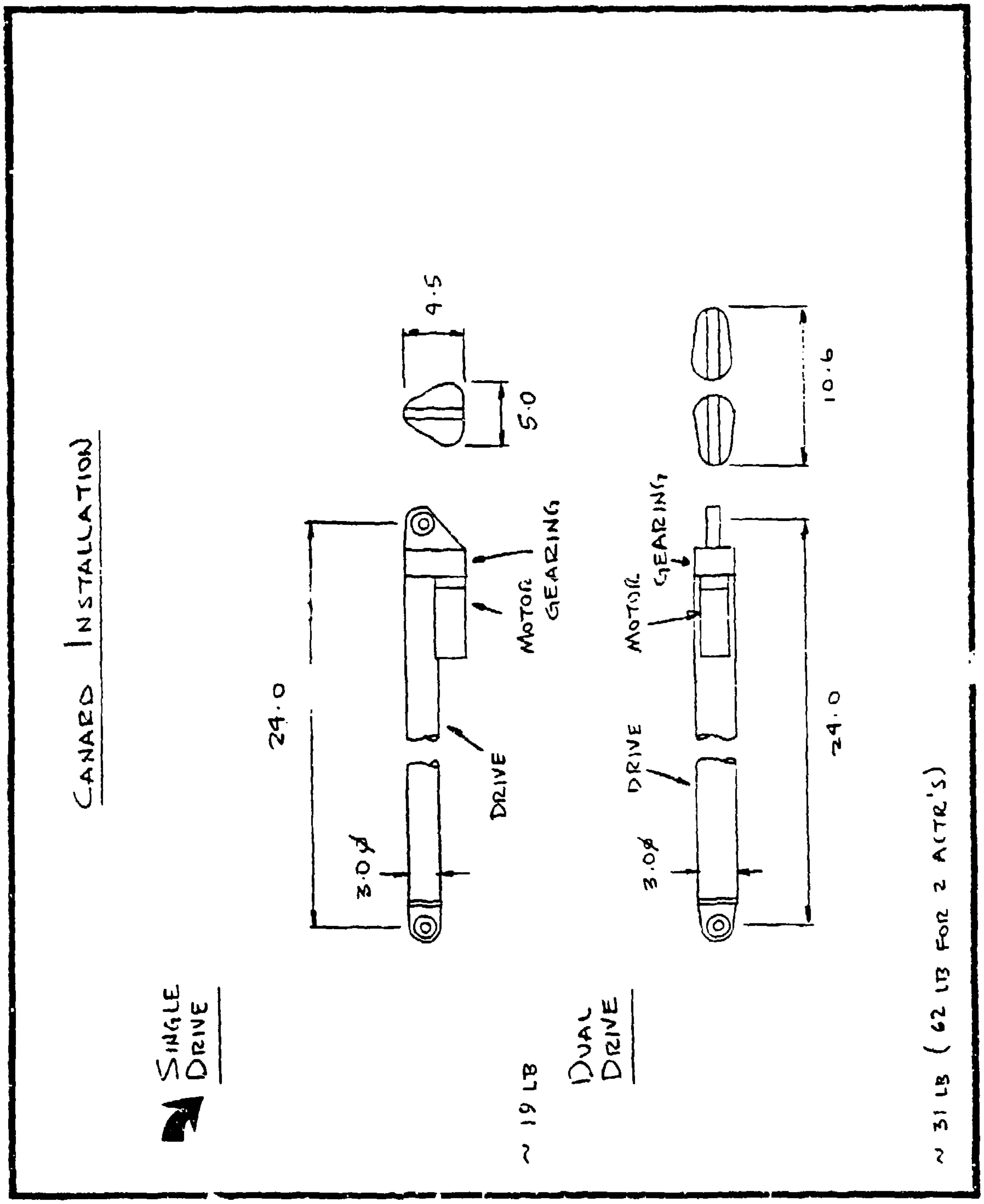
4. CLUTCH
   - DIAMETER ~ 2.3 IN
   - LENGTH ~ 1.0 IN
   - WEIGHT ~ 1.2 LB

5. ROD ENDS
   - WEIGHT 2.0 LB

DATE 9-22-80 | Appendix C | CALC. NO. 2-007 SHEET NO. 2

PART NO. TEAATS | MODEL NO.

PREPARED BY S. ROWE | CHECKED BY



Appendix C

Inverter Current Requirements

DATE 10-16-80
PART NO.
PREPARED BY S. ROWE

Appendix C

CALC. NO. 3-001 SHEET NO. 1
MODEL NO.
CHECKED BY

APPROXIMATE MOTOR CURRENT RATINGS & INVERTER SELECTIONS

	ACTUATOR	MOTOR SPEED	MOTOR POWER	VOLTAGE DROP	η_m	MOTOR CURRENT	INVERTER RATING
①	INBOARD FLAP	20KRPM	33.6 KW	240 V	0.9	156 A	150 A *
②	MIDSPAN FLAP	20KRPM	6.7 KW	240 V	0.9	31 A	50 A *
③	AILERON	20KRPM	3.0 KW	240 V	0.9	14 A	20 A **
④	UPPER RUDDER	20KRPM	0.75 KW	240 V	0.9	3.5 A	5 A **
⑤	LOWER RUDDER	10KRPM	1.5 KW	240 V	0.9	6.9 A	10 A **
⑥	L.E. FLAP	10KRPM	3.7 KW	240 V	0.9	17 A	20 A **
⑦	CANARD	10KRPM	0.75 KW	240 V	0.9	3.5 A	5 A **

* USE INVERTERS DESIGNED IN AIR 80-17284

** REQUIRES 25 AMP OR SMALLER

Appendix D-1

Inverter Loss Summary

DATE 1-8-80 Appendix C CALC. NO. 3-005 SHEET NO. 1

PART NO. MODEL NO.

PREPARED BY S. ROWE CHECKED BY

INVERTER LOSSES FOR 25% DUTY CYCLE *

ACTUATOR	MOTOR CURRENT	INVERTER TYPE & RATING	25% CURRENT	INVERTER LOSSES	TOTAL INVERTERS	TOTAL LOSSES
INBOARD FLAP	156 A	EC ** 150 A	39 A	316.9 W	6	1901 W
MIDSPAN FLAP	31	EC ** 50	7.8	56.9	6	341
AILERON	14	NC 20	3.5	6.6	4	26
UPPER RUDDER	3.5	NC 5	0.9	1.6	6	10
LOWER RUDDER	6.9	NC 10	1.7	3.0	6	18
LE FLAP	17	NC 20	4.3	8.7	12	104
CANARD	3.5	NC 5	0.9	1.5	4	6
				Σ LOSSES =		2406 W

* DATA FOR 50% SPEED
** DATA FROM A.R. REPORT 80-17284

DATE 1-9-83 Appendix C CALC. NO. 3.005 SHEET NO. 2

PART NO. MODEL NO.

PREPARED BY S. ROWE CHECKED BY

INVERTER NOMENCLATURE

EC	EVAPORATIVE COOLING
FC	FORCED CONVECTION
NC	NATURAL CONVECTION

Appendix D-2

Natural Radiation and Convection Analysis

DATE 1/15/81 Appendix C CALC. NO. 3-006 SHEET NO. 1

PART NO. MODEL NO.

PREPARED BY R. MILO CHECKED BY

AATS / ROCKWELL - INVERTER COOLING ANALYSIS

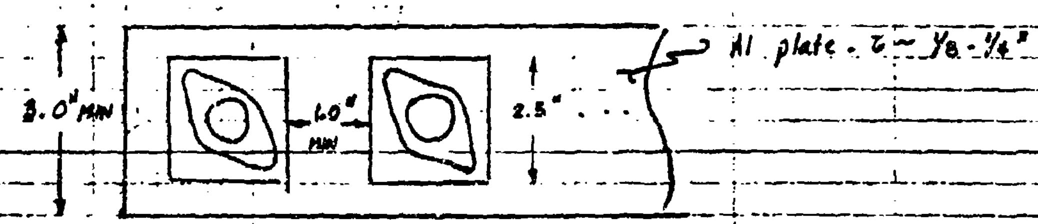
TEMPERATURE PREDICTIONS FOR LE FLAP INVERTERS :

DUTY CYCLE	⊛ POWER per TO3 DEVICE	TEMP ~ °C CASE	JUNCTION	Notes
1. SPEED ~ 50% LOAD ~ 25% TIME ~ ∞	1.8 W	127	128	θ_{j-c} = 0.42 °C/W
2. SPEED ~ 50% LOAD ~ 50% ⊛ TIME ~ 120 SEC	4.85 W	137	139	θ_{j-c} = 0.42 °C/W INITIAL CASE TEMP, T_{c_i} = 127 °C

⊛ 6 circuits (TO3's) per Inverter

⊛ Transient Temp Response after Duty Cycle 1 with elapsed time of 120 seconds

- HEAT SINK / DISSIPATOR (ATTACHED) : IERC PN. HP1-TO3-33B
- COOLING APPROACH : NAT. CONV. + RADIATION To 107 °C Ambient



Assumption : 1) No Insulator between TO-3 and Heat Sink

2) 107 °C Ambient

3) θ_{j-c} = 0.42 °C/W

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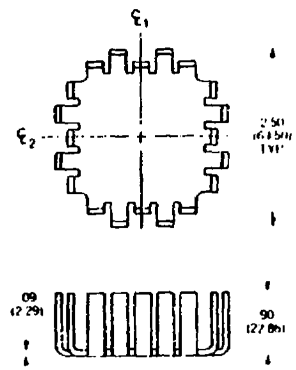
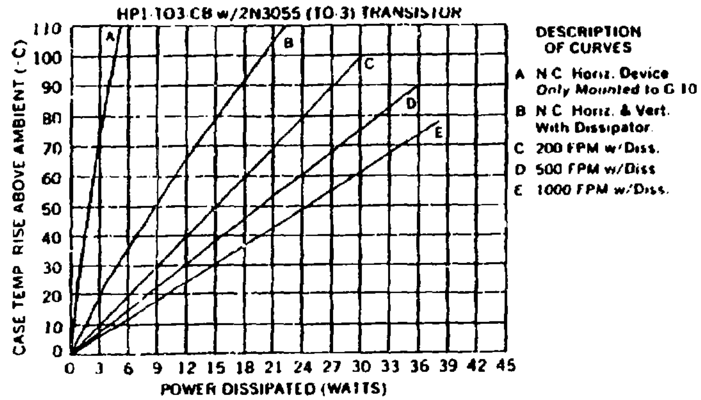
DATE 1/15/81 Appendix C CALC. NO. 3-006 SHEET NO. 2
PART NO. MODEL NO.
PREPARED BY R. Milo CHECKED BY

IERC

HP1 Series for Single TO-3 or Stud Mount Devices



Dimensions are for reference use only. Contact IERC for dimensions with tolerances or standard part drawings



- Thermal Resistance Case to Sink is 0.1-0.3 °C W w/ Joint Compound
- Derate 1.0 °C watt for unplated part in natural convection only
- Derate 0.4 °C watt for Insulube* part in natural convection only.

Ordering Information

IERC PART NO						
Unplated	Comm'l Black Anodize	Mil Black Anodize	Insulube* 448	Semiconductor Accommodated	Hole patt ref no. (see pg 1-28)	Max Weight (Grams)
HP1-000 U	HP1-000 CB	HP1-000 B	HP1-000	Undrilled		35.0
HP1-103 U	HP1-103 CB	HP1-103 B	HP1-103	TO-3	16	35.0
HP1-103-33U	HP1-103-33CB	HP1-103-33B	HP1-103-33	TO-3 IC	17	35.0
HP1-103-44U	HP1-103-44CB	HP1-103-44B	HP1-103-44	TO-3 panel mount	31	35.0
HP1-436 U	HP1-436 CB	HP1-436 B	HP1-436	TO-3 (4 pin)	18	35.0
HP1-106 U	HP1-106 CB	HP1-106 B	HP1-106	TO-6, TO-36	19	35.0
HP1-1015 U	HP1-1015 CB	HP1-1015 B	HP1-1015	TO-15	23	35.0
HP1-420 U	HP1-420 CB	HP1-420 B	HP1-420	Universal	27	35.0

Note: See page iv for other finishes.

Appendix D-3

Evaporative Cooling Analysis

Appendix D-3

Evaporative Cooling Analysis

Evaporative Cooled Inverter Analysis

As a baseline configuration, a 50-amp inverter with a 50-percent motor speed and current duty cycle was used for analysis [1]. At this current rating, the inverter bridge consists of six electronic circuit packages, each with a FET in a TO-3 can that dissipates approximately 50 watts. The cooling requirement is to maintain the FET junction temperature below 150°C; preferably below 125°C.

R-113 (trichlorotrifluoroethane) was selected as the coolant. It has been used successfully for evaporative cooling of power semiconductors for traction motor choppers. The boiling heat transfer coefficients for R-113 were derived from recent AiResearch heat transfer test results for a thyristor/busbar assembly immersed in R-11 (trichlorofluoromethane) [2].

A cylindrical configuration was used for the inverter. The six electronic circuit packages would be contained in a right circular cylinder which is filled approximately by 90-percent in volume with R-113. The cylindrical container serves as the R-113 condenser. When boiling occurs, a vapor zone above the liquid is generated and condensation takes place on the metal surface of the container and the vapor returns the liquid state. As such, the container/condenser should be designed to meet both inverter packaging and heat transfer requirements. The R-113 temperature selected for the baseline condition was 93.3°C (200°F). The corresponding vapor pressure is 54.7 psia.

Forced air cooling of the inverter assembly was selected due to simplicity. F-1 shows a schematic of the forced convection cooling approach. AiResearch plate-fin matrices are placed over the hollow cylindrical inverter and 130°F air is blown through the fin passages keeping the inside metal surface relatively cool for R-113 condensation. Also shown in F-1 is the effect of air volumetric flow on the overall size of the inverter and on the resultant pressure drop across the plate-fin matrices. With a fixed air flow, the plate fin configuration (12R-0.3-0.5(0)-0.006(al))* indicated high heat transfer capability while keeping the pressure drop within acceptable range. To acquire a low fan energy consumption, the pressure drop which is a square function of the air flow should be kept at a minimum. The design air flow of 50 cfm exhibits this characteristic depicting a pressure drop of 2 inches of H_2O. An incremental reduction of the air flow will significantly increase the inverter size as projected in F-1. An increase in air flow results in a substantial increase in the pressure drop.

A diameter of 5 inches (over fins) was chosen for the 50-amp inverter. The data of Figure C-6 reflects this choice. A length of 8 inches for the heat exchanger provided the required surface area. Table C-5 provides a summary of pertinent data.

Dimensions for alternate ratios of length to diameter may be calculated by maintaining a constant heat exchanger surface area. Also, the unit may be scaled for larger current ratings by holding the dissipated watt-inch^{-1}constant, and maintaining boundry conditions.

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*AiResearch heat exchanger designation.

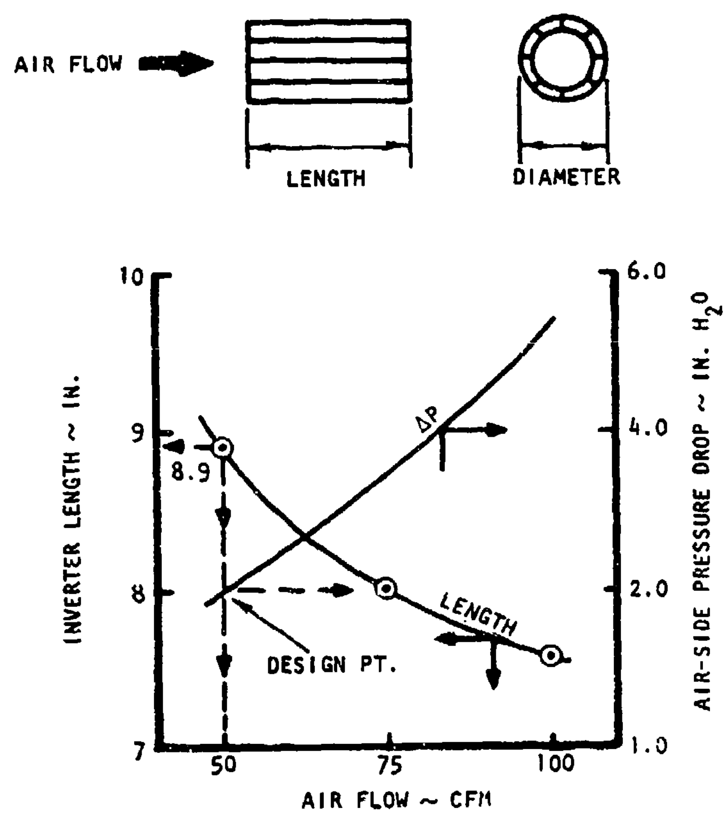


Figure C-6. Effect of Air Flow on Inverter Size

Table C-5

50-Amp Inverter Heat Exchanger Summary

• Diameter	5.0 in
• Length	8.0 in
• Configuration	Aluminum, radial fins
• Coolant Data	
Type	R-113
Temperature	200°F
Pressure	54.66 psia at 200°F
Density	97.45 $lb\text{-}ft^{-3}$ at 80°F
Volume	90 percent fill
• Cooling	
Type	Air
Temperature	130°F
Flow	50 cfm

References

1. Rowe - AIResearch Report No. 80-17284, Electromechanical Airplane Actuation Trade Study

2. Kim - AIResearch Document No. 09303-46685-D11, "Pole Face - to - R-11 Thermal Resistance - Evaporative Cooling of Power Semiconductors for Rail Vehicle Choppers"

APPENDIX D

AIRPLANE ACTUATION TRADE STUDY

This appendix includes the work done by AiResearch, under the direction of S. Rowe as part of their effort, in accordance with service agreement L9FM-11231-405, to provide a detailed design of the inboard flap actuation system. This detailed design was undertaken to give a higher level of confidence in the projected weights, envelopes and life cycle cost figures used for the other 6 flight control actuators. The data generated in this appendix also provided a basis for the motor weights shown in Figure 38.

ALL ELECTRIC AIRPLANE STUDY
SERVICE AGREEMENT L9FM-11231-405
PROGRESS REPORT FOR THE PERIOD FEBRUARY - MARCH 1981: ADVANCED ELECTROMECHANICAL ACTUATION SYSTEMS

PREFACE

This document is submitted in conjunction with Service Agreement L9FM-11231-405, All Electric Airplane Study. The data submitted herein is presented to facilitate completion of the subject study. This document, in addition to previously submitted data, shall serve as a progress report for February - March 1981.

INTRODUCTION

Specific tasks completed during this reporting period were:

- Inboard flap actuation system detailed design
- Motor performance trades based on 1990 technology projections
- Study overview and summary

Each of the above is addressed in this progress report.

This submittal satisfies the statement-of-work (SOW) data requirements, and completes the subject study.

DETAILED DESIGN

Detailed design of the inboard flap actuation system was performed in accordance with the SOW [1]. The inboard flap was selected by mutual agreement between the customer and AiResearch.

The completed design is described in the following drawings:

Drawing No.	Title
2022824	System Outline, Flight Control
2022794	Gearbox Outline, Flight Control
2022796	Gearbox Outline, Flight Control
2022798	Power Drive Unit Outline

Performance data for the system is tabulated in Table D-1. System weights are shown in Table D-2, including the required motor inverters and a system controller. Additional system and component data may be found on the drawings.

Table D-1

Inboard Flap Performance

Stroke	+30°
	-45°
Rate	100 deg-sec^{-1}
Load	453 X 10^3 in-lb
Bandwidth (+1°)	> 30 rad-sec^{-1} (at 0 dB)
Static Stiffness	58 X 10^6 in-lb

Table D-2

Inboard Flap Weight Summary

Controller:	1 X 5 lb =	5 lb
Inverter :	3 X 38 =	114
Actuator :	1 X 100 =	100
Position Feedback:	1 X 1 =	1
		220 lb

A brief description of the actuation system and its components is given below.

Actuation System Description (2022824)

The inboard flap actuation system was designed to satisfy the operating and performance requirements determined by the customer and AiResearch [2]. The actuation system consists of the following components:

Item	Balloon No.	Qty	Description
1	-	1	Controller
2	-	3	Inverter
3	1	1	Power Drive Unit
4	2	1	Torque Tube
5	3	1	Reduction Gearbox
6	4	1	Torque Tube
7	5	1	Hingeline Gearbox
8	-	1	Position Feedback Assembly

Items 1 and 2 have been addressed in previous reports [1, 2, 3]. Item 8 was assumed to be a module containing three RVDTs and reduction gearing, but was not addressed in detail.

Operation of the actuation system is typical of 270 vdc systems (see Figure D-1 and drawing No. 2022824). The controller commands the inverters as a function of servo position and motor speed feedback. Each inverter in turn controls the current to its respective motor in the power drive unit (PDU) by means of a current feedback loop. Motor response is torque summed in the PDU, and is output through a torque tube at motor speed (1:1 ratio). This torque tube then drives a reduction gearbox (88:1 ratio) and makes a 90° turn, so that the reduction gearbox output lies on the hingeline gearbox centerline. The hingeline gearbox (15:1 ratio) then positions the inboard flap. Feedback loops are provided at the reduction gearbox (actuator position), PDU (motor speed and rotor position), and inverters (motor current).

Actuation system components are described in the following paragraphs.

Power Drive Unit (2022798)

Each PDU consists of three brushless direct current-permanent magnet (DC-PM) servomotors, each powered from a 270 vdc source via inverters. Motor torque is output through clutches (which can decouple a failed motor from the actuator drive) to a torque summing gear train. PDU output is at 90° to the motor axes due to actuation system installation. All gearing is supported on ball bearing assemblies. Due to motor/gearing speeds (max speed is 22 Krpm), oil sling lubrication is used in the PDU gear housing.

Additional data on motor design may be found in drawing 2022798, and in a later section of this report. Additional data on the PDU gearing is presented in the drawing, also.

Reduction Gearbox (2022796)

The reduction gearbox provides speed reduction between PDU and the hingeline gearbox. It consists of compound planetary gearing supported on roller bearing assemblies; and a bevel gear at the input supported on ball bearing assemblies. Position feedback for the actuation system is provided by an output shaft which drives a redundant RVDT feedback module.

See drawing 2022796 for additional information.

Hingeline Gearbox (2022794)

The hingeline gearbox comprises fourteen identical stages ("slices") of compound planetary gearing operating in mechanical parallel. The load is distributed uniformly (approximately) along the length of the gearbox. An input shaft, which runs the full length of the gearbox, is supported on ball bearing assemblies and engages planets at each slice. The planets then drive each output slice, positioning the inboard flap.

Drawing 2022794 provides additional data on the hingeline gearbox.

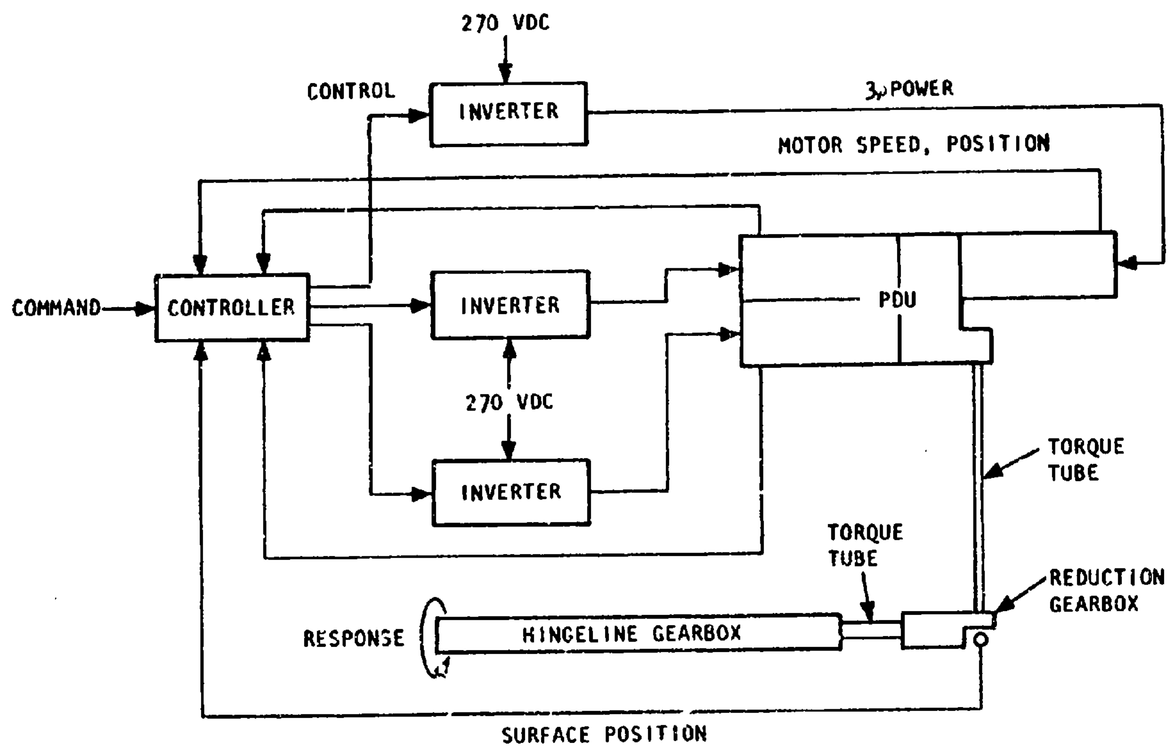


Figure D-1. Inboard Flap Actuation System Block Diagram

Comments and Recommendations

The design presented in this report for the inboard flap meets or exceeds the operating and design requirements agreed upon by the customer and AiResearch. This has incurred a penalty in terms of system weight, however.

Referring to Table D-1, it is noted that the maximum operating load (stall) of the actuation system is 453 X 10^3 in.-lb, while the maximum load seen by the inboard flap is 216 X 10^3 in.-lb [4]. This results from the design stall torque of the three torque summed motors. The drive of the actuation system, including the hingeline gearbox, was designed to accommodate this stalled condition. If an acceptable assumption is that the maximum drive load is 216 X 10^3 in.-lb under any circumstance, then system weight could be reduced. For example, the hingeline gearbox weight would drop from 50 lb to approximately 25 lb if this assumption were enforced.

It is recommended that the customer evaluate the feasibility of this design assumption.

MOTOR PERFORMANCE TRADES

Motor performance trades using assumed 1990 technology were performed early in the study. Data resulting from this effort is presented in Appendix A.

All motors used in actuation systems sized or designed during the study were selected in accordance with the criteria of Appendix A.

STUDY OVERVIEW AND SUMMARY

This section presents an overview of the work accomplished during the study, in chronological order; and a summary of study findings.

Overview

The study SOW [1] was reviewed during September 1980 and work initiated during that same month. Trades with particular emphasis placed on actuation system configuration and control were performed and used to establish the approach to be utilized [3].

From October 1980 through January 1981 the customer's design requirements were evaluated [4]; and preliminary designs for specific actuation system and related components were completed [2, 5].

February through March 1981 saw the selection of the inboard flap actuation system for in-depth analaysis, and completion of its detailed design [5].

Summary

Conclusions drawn from the study program are presented below:

1. Motor designs investigated provided low weight designs with excellent dynamic response capability. For the environment and duty cycle considered, active cooling was found to be unnecessary.

2. Inverter designs are driven by thermal considerations. For the device characteristics and duty cycle assumed, evaporative cooling appears to be a likely choice for high current units (25 amps or more); while low current units (25 amps or less) were adequately cooled by means of conventional heat sinking.

3. Actuation system control is best accomplished by a microprocessor based controller for surface position, motor speed, and rotor position. Due to required calculation rates, use of digital control for the current loop is impractical and will remain analog.

4. Available envelopes for the actuation systems played an important role in establishing actuation system configurations. In most cases, PDUs driving hingeline rotary gearboxes were found to be most suitable.

5. Actuation system weight in most cases can be reduced by sizing the mechanical drives for maximum design loads, as opposed to simultaneous motor stall. The acceptability of this approach would require review by the customer.

REFERENCES

1. Green -- AiResearch Letter No. CBRG:9327:1011, Subject: Rockwell Service Agreement, L9FM-11231-405, All Electric Airplane Study.

2. Rowe -- AiResearch Report No. 80-17351 (2), All Electric Airplane Study, Progress Report for the Period October 1980 - January 1981.

3. Rowe -- AiResearch Report No. 80-17351 (1), All Electric Airplane Study, Progress Report for the Month of September.

4. Helsley -- Rockwell International Report No. NA-79-492, Airplane Actuation Trade Study, First Interim Technical Report.

5. Rowe -- AiResearch Report No. 80-17351 (3), All Electric Airplane Study, Progress Report for the Period February - March 1981.

APPENDIX A

MOTOR SIZING, ANALYSIS AND COMPUTER PROGRAMS

APPENDIX A-1

MOTOR SIZING AND ANALYSIS

Motor selection is probably the most critical phase of EM actuation system design. The motor selection will determine actuator performance, influence actuator configuration, and dictate actuation system control. Design constraints placed on the motor will affect the actuation system design, also.

For primary flight control servo applications, brushless DC-PM (direct current-permanent magnet) motors appear to be the most likely motor candidate [1, 2]. A 270 to 300 vdc power source is probable, due to the availability of 400 Hz 3-phase ac power and past experience with high voltage dc actuation systems [1, 3, 4]. Servo control would be accomplished by means of a transistorized inverter, with control logic for inverter switching (commutation), current limiting, and control law implementation. Figure D-2 illustrates this type of machine.

For other servo applications, induction motors and brush DC-PM motors are likely candidates. Advancements of these types of motors were believed to be of less importance than the brushless DC-PM motor for high performance servo applications and thus not considered for trade studies.

A.1 ASSUMPTIONS

Assumptions defining technology advancements and motor configuration were made, and used as a starting point for motor trades. The assumptions developed were believed to be realistic and reasonable, for the time frame involved. Assumptions with rationale are provided below.

A.1.1. Magnets

Determination of magnet energy product and characteristics are significant tasks in motor design. Materials with energy products as high as 30 x 10^6 gauss-oersted have been developed, and materials with energy products of 23-26 x 10^6 gauss-oersted are readily available [5, 6, 16]. However, most magnets are presently supplied in the 16-22 x 10^6 gauss-oersted range. While increased energy product is generally desirable for motor design, other factors must be considered in material selection.

Coercive force (H) must be sufficiently large in magnitude to allow full utilization of the material's flux density (B). Insufficient coercive force could allow demagnetization due to motor currents beyond design limits (short circuit, current limit overshoot, etc.). Additionally, elevated operating temperatures decrease magnet energy product, and susceptibility to demagnitization increases. These effects are illustrated by the B-H plots shown in Figure D-3.

Increased magnet energy product (23-30 x 10^6 gauss-oersted) is also associated with increased cost. This is due primarily to the need for significant fractions of the less common rare earths and limited source metals (samarium, praseodymium, and cobalt). Increased demand for these materials would affect cost and availability in the future [17].

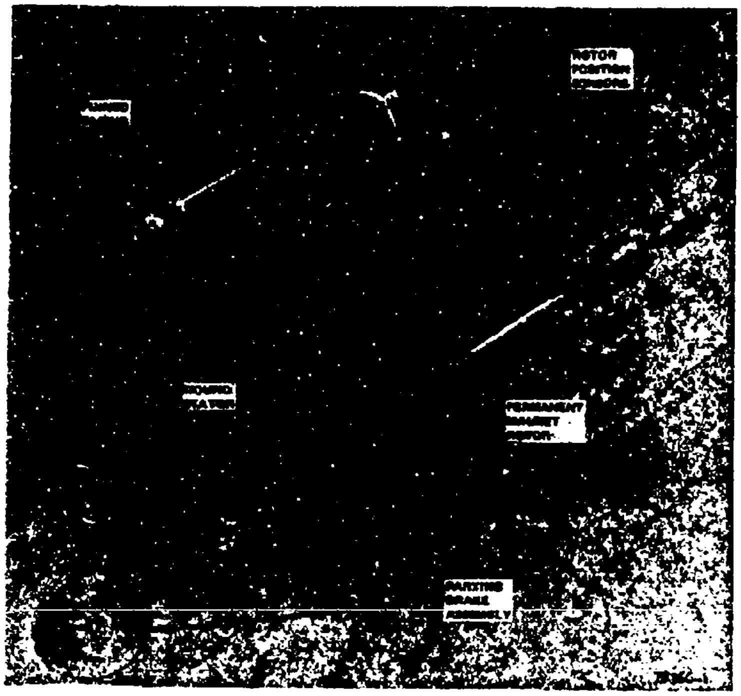


Figure D-2. Brushless DC-PM Motor

F-33923

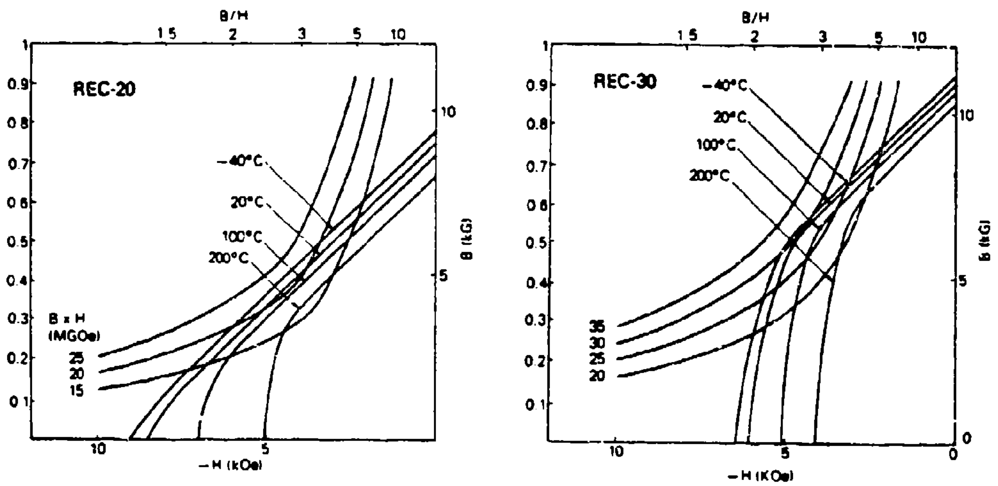


Figure D-3. Comparison of Magnet Materials [16]

The number of manufacturers producing high-energy product magnets is quite limited. Most deal in the 16-22 x 10^6 gauss-oersted range. This factor would tend to restrict future supplies, also.

Recent trends in rare earth-permanent magnet development emphasize reduced cost and improved supply rather than increased energy product [17]. In light of the aforementioned factors, the probability of using very high energy product magnets appears to be unlikely. Rather, use of new magnet materials with more desirable characteristics in the 20-24 x 10^6 gauss-oersted range appears likely for the near term.

For these reasons an energy product of 22 x 10^6 gauss-oersted was used as a reasonable compromise. This energy product represents a balance between B, H, cost, and availability for rare earth-permanent magnets during the near term.

It should be noted for the operating temperatures anticipated (250°F ambient) that any difference in performance between machines using 22 x 10^6 and existing 30 x 10^6 gauss-oersted materials would be negligible due to the degraded performance of the highest energy product materials at elevated temperature.

A.1.2 Motor Speed

A maximum motor no-load speed of 25 Krpm was assumed as a design constraint, for reasons explained later. Running motors at the highest speed possible is desirable since this reduces motor size and weight for a fixed output power.

The following relations define motor power, speed, and torque:

$$\dot{W}_m = T_m \dot{\theta}_m \tag{A-1}$$

$$T_m = K_t I_m \tag{A-2}$$

$$\dot{\theta}_m = K_e^{-1} V_m \tag{A-3}$$

where

$\dot{W}_m$ = motor power

T_m = motor torque

$\dot{\theta}_m$ = motor speed

I_m = motor current

V_m = motor voltage drop

K_t = torque constant

K_e = voltage constant

Increasing motor speed will decrease motor copper loss* due to winding changes, allowing reduced machine dimensions. Thermal effects are usually a constraint in motor dimensions.

Motor speed cannot be increased indefinitely, however, due to motor and actuator design constraints. Hoop stress in the rotor magnet sleeve due to centrifugal loads must be considered, as well as rotor surface velocity. In electrical machines, stator hysteresis losses and eddy currents can become excessive as frequency increases. Inverter switching frequency and switching losses may increase. Additional drive reduction is required at high speeds, for a fixed output speed, and gearing is usually velocity limited to 30 Krpm.

Thus, an upper limit of 25 Krpm was used as a reasonable compromise, based on design constraints and past experience.

A.1.3 Dynamic Response

Since the motors were meant for use in primary flight control servos, frequency response was a prime consideration as a performance parameter. Most EM actuation systems are acceleration limited, thus determining the maximum possible bandwidth.**

Bandwidth (no-load) for a single motor servoactuator, assuming acceleration saturation, may be calculated from

$$\omega = \sqrt{\frac{\ddot{\theta}_m}{A \times G}} \tag{A-4}$$

$$\ddot{\theta}_m = \frac{T_m}{J_{eff}} \tag{A-5}$$

$$T_m = K_T I_m \tag{A-6}$$

$$J_{eff} = J_m + \frac{1}{G^2 J_l} \tag{A-7}$$

where

ω = bandwidth

$\ddot{\theta}_m$ = motor acceleration

T_m = motor torque

I_m = motor current

* Winding resistance losses; see paragraph A.1.4.

**This is true for operation as a linear system; i.e., up to acceleration saturation. Bandwidth for the system in the nonlinear (saturated) region may be acceptable depending upon the frequency response specification.

J_m = motor inertia

J_l = load inertia

J_{eff} = effective inertia, reflected to motor

K_t = torque constant

A = actuator position command amplitude

G = actuator gear ratio

η = gearing efficiency

Equations (A-4) through (A-7) illustrate that there are numerous possible solutions for motor and actuator parameters (I_m, K_T, J_m, G, η) given ω, A, and J_l. Attempts have been made to develop analytical procedures for "optimum" motor design based on bandwidth requirements; but they failed to address other motor and actuator constraints [1, 7, 8, 9].

In order to develop a family of motors which would provide adequate bandwidth for the various actuation systems under consideration, the following criteria were employed:

$$\tau_m = \frac{\dot{\theta}_{rated}}{\ddot{\theta}_{stall}} \simeq 50 \text{ msec} \qquad \text{(A-8)}$$

$$\frac{l}{d} = \frac{l}{2r} \leq 3 \qquad \text{(A-9)}$$

where

$\dot{\theta}_{rated}$ = motor speed at rated power

$\ddot{\theta}_{stall}$ = motor acceleration at stall torque

τ_m = motor time constant

d = motor rotor diameter

l = motor rotor length

r = motor rotor radius

Figure D-4 illustrates motor dimensional parameters; Figure D-5 illustrates performance parameters.

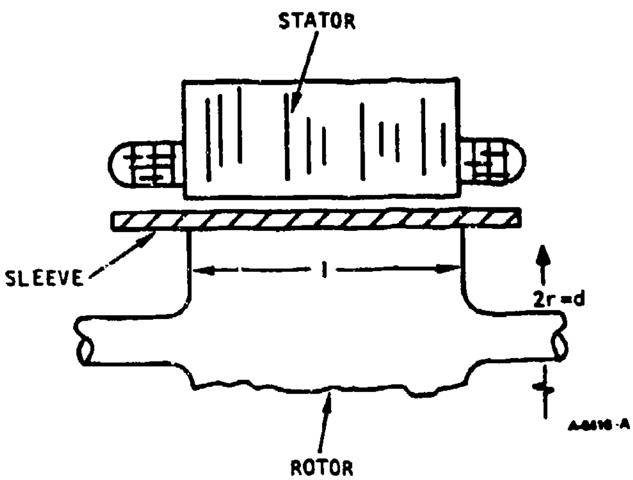


Figure D-4. Brushless DC-PM Motor Detail

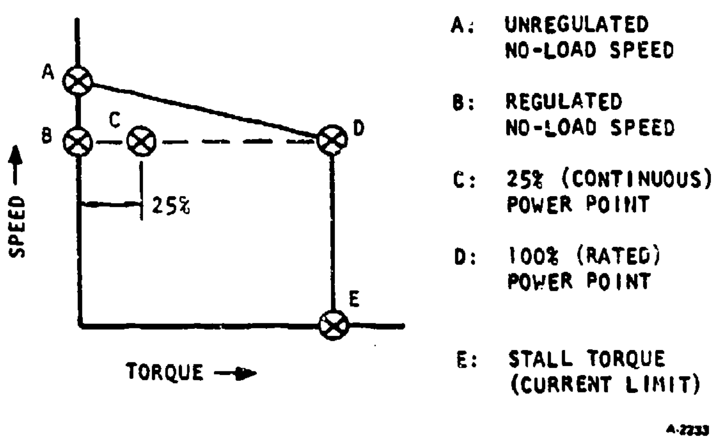


Figure D-5. Motor Performance Points

τ_m is the time required for the motor to accelerate from Point E to Point D of Figure D-5. The motor accelerates under constant torque with no-load, which is accomplished via current limit control in the inverter. This parameter is a measure of the motor's dynamic response; it should not be confused with the time constant which reflects the time required for a motor to reach 63.2 percent of commanded speed for a step command, i.e.,

$$\dot{\theta}_m (t) = \left[\frac{V_m}{K_e}\right][1 - \exp(-t/\tau)] \qquad \text{(A-10)}$$

$$\tau = \frac{J_m R}{K_t K_e} \qquad \text{(A-11)}$$

where R is the motor winding resistance, t is time, and V_m is a step voltage applied across the motor. The above time constant results from a linear analysis assuming voltage control of the motor, and does not assume constant torque (constant current) operation [10].*

Equation (A-9) is a dimensional constraint owed to manufacturing difficulties and dynamic stability. Generally, if the rotor l/d ratio exceeds 3:1, winding the motor stator becomes difficult due to motor configuration (see Figure D-6); dynamic stability of the rotor becomes a concern due to bending modes. Also a large l/d ratio favors lower values of τ_m, since this produces a lower rotor inertia than a small l/d.

Utilizing the two constraints, τ_m and l/d, allowed a family of motors to be designed without regard to detail actuator performance characteristics. Since the motors were expected to satisfy or exceed dynamic response requirements for most actuator applications, motor characteristics could be parametrically tabulated for trend analysis. Also, motors could be selected by use of parameters other than dynamic response, greatly simplifying the design process.

A.1.4 Thermal Considerations

The performance of any motor is limited by duty cycle. Losses developed by the motor as a function of load and speed must be considered during motor design (and during motor selection), or overheating of the windings may occur.

The two most significant losses which must be accounted for are resistance (copper) losses, and hysteresis and eddy current (iron) losses. Relations for these losses are [11]:

*The analysis used (equations A-10 and A-11) develops the motor mechanical time constant for the case of negligible inductance and viscous losses.

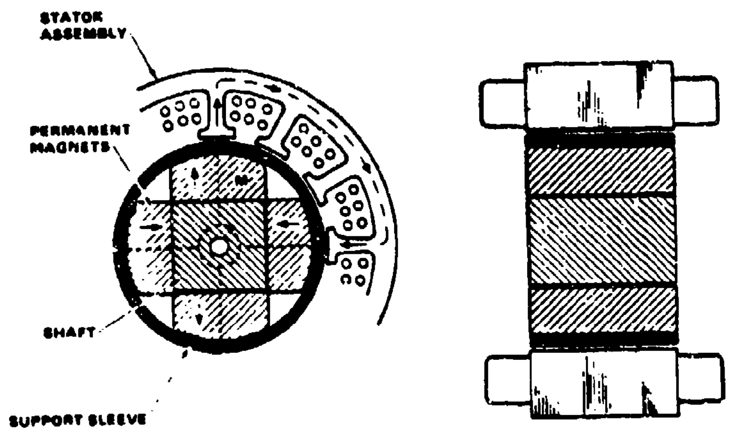


Figure D-6. Stator Cross Section

$$\dot{W}_{cu} = I_m^2 R \qquad \text{(A-12)}$$

$$\dot{W}_{ir} \propto \omega_m K_h + \omega_m^2 K_{ec} \qquad \text{(A-13)}$$

where $\dot{W}_{cu}$ = copper loss

$\dot{W}_{ir}$ = iron loss

I_m = motor current

ω_m = motor (commutation) frequency

R = winding resistance

K_{ec} = eddy current loss coefficient

K_h = hysteresis loss coefficient

Iron losses are usually controllable by judicious machine construction and material selection; copper losses may be difficult to control depending upon stator resistance and duty cycle. In order to maintain manageable steady-state and transient temperatures, a maximum current density of 18,000 amp per square inch of conductor was chosen. The selection of this value was based upon analysis of the specification duty cycle (see paragraph A.2).

A.1.5 Other Assumptions

Other assumptions for motor design included a six-pole rotor configuration, with radially oriented magnets. Additional constraints and assumptions were made, consistant with experience.

Table D-3 summarizes the principal design assumptions and constraints.

TABLE D-3

MOTOR DESIGN ASSUMPTIONS AND CONSTRAINTS

Magnets:	$BH = 22 \times 10^6$ gauss-oersted
No-Load Speed:	$\dot{\theta}_{max} \leq 25$ Krpm
Time Constant:	$\tau_m \leq 50$ msec
Rotor Dimensions:	$l/d \leq 3$
Stator Current Density:	18 Kamp-in^{-2}
Rotor Configuration:	6 pole, radial magnet orientation

A.2 DESIGN INVESTIGATION

A family of motors using the criteria of Table D-3 was designed by use of a computer program developed at AiResearch [12]. A brief description of the program is given in Appendix A-2.

Motors were designed as a function of peak (rated) power, over a range of horsepower. Key motor data are tabulated in Table D-4. Some of the data were used to calculate parameters of interest and plotted, also. These data are shown in Figures D-7 and D-8.

Since motor dynamic response was a key parameter in selecting motor design constraints, a more detailed investigation of τ_m and its relation to other motor parameters was conducted. Two general cases were examined: (1) motor characteristics as a function of l/d for fixed rated speed and τ_m; and (2) motor characteristics as a function of τ_m for fixed $\dot{W}_m$ and l/d.

Four values of l/d were investigated for the first case. A rated load speed of 20 Krpm was used. Data for this case are tabulated and plotted in Table D-5 and Figure D-9, respectively.

In the second case, three values of τ_m were run, assuming a motor power of 60 hp. This power rating was used since τ_m can cause extreme variations in motor size and weight for larger machines. Data for this case are presented in Table D-6 and Figure D-10.

TABLE D-4

MOTOR DATA SUMMARY

Parameter	Rated Power (hp)						
	1	5	10	15	20	25	30
Housing Diameter (in.)	2.0	2.5	2.7	2.9	2.9	3.1	3.2
Housing Length (in.)	4.7	5.9	7.0	8.0	8.9	9.4	9.6
Motor Weight (lb)	1.0	2.0	3.5	5.0	6.5	9.5	10.5
Current Limit (amp)	3.9	17.9	35.4	53.1	70.8	90.0	106.3
Thermal Flux Density (watt-in.^{2})*							
• 25 percent	0.83	1.05	1.14	1.21	1.24	1.24	1.32
• 100 percent	13.3	16.8	18.3	19.3	20.0	20.0	21.1
Speed (krpm)**							
• A	25.6	23.9	23.5	21.0	21.0	16.5	15.1
• B, D	20.0	20.0	20.0	18.0	18.0	16.0	15.0
Torque (in.-lb)**							
• D, E	3.1	15.7	31.7	52.8	70.5	98.8	126.6
Time Constant (msec)	46.3	48.0	49.7	49.7	49.8	50.0	50.0

* Copper loss only.
** See Figure A-4.

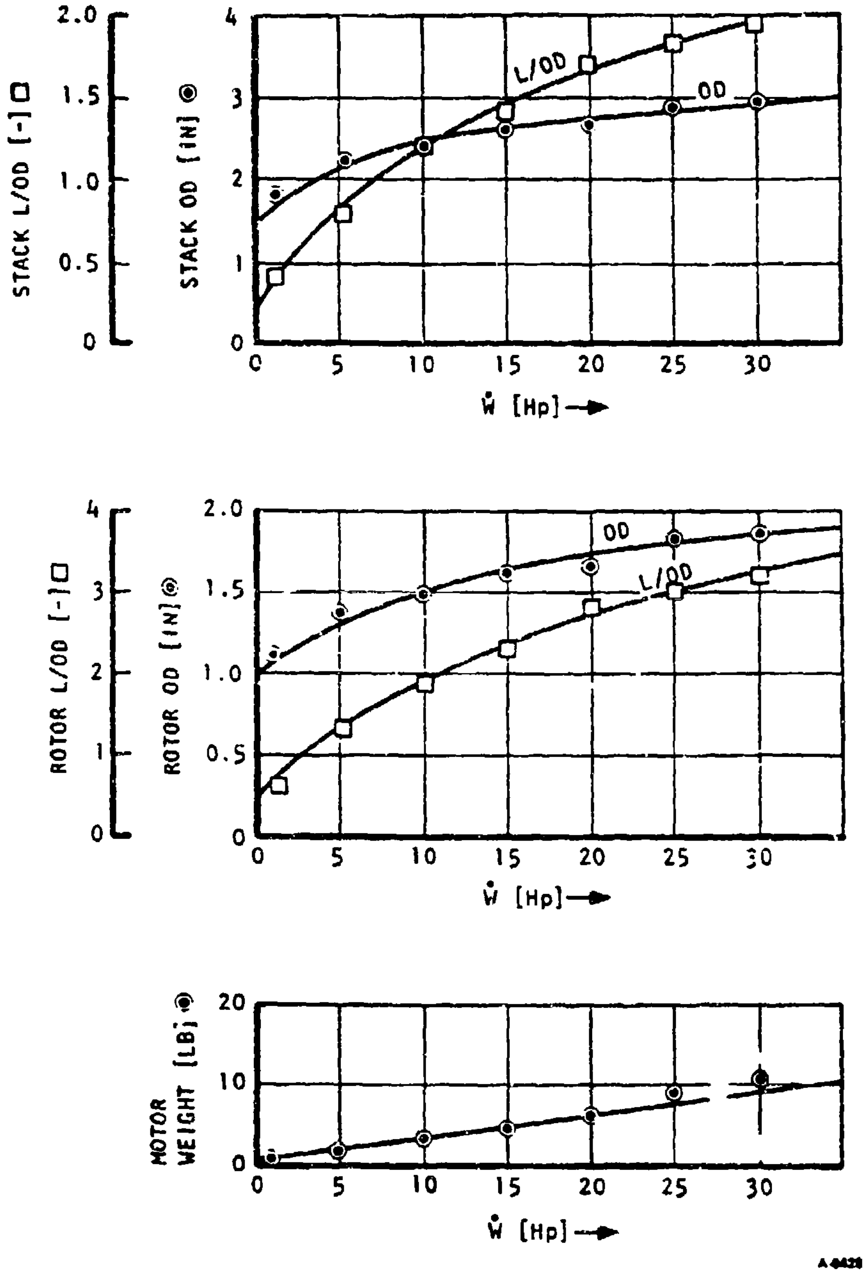


Figure D-7. Motor Data Summary

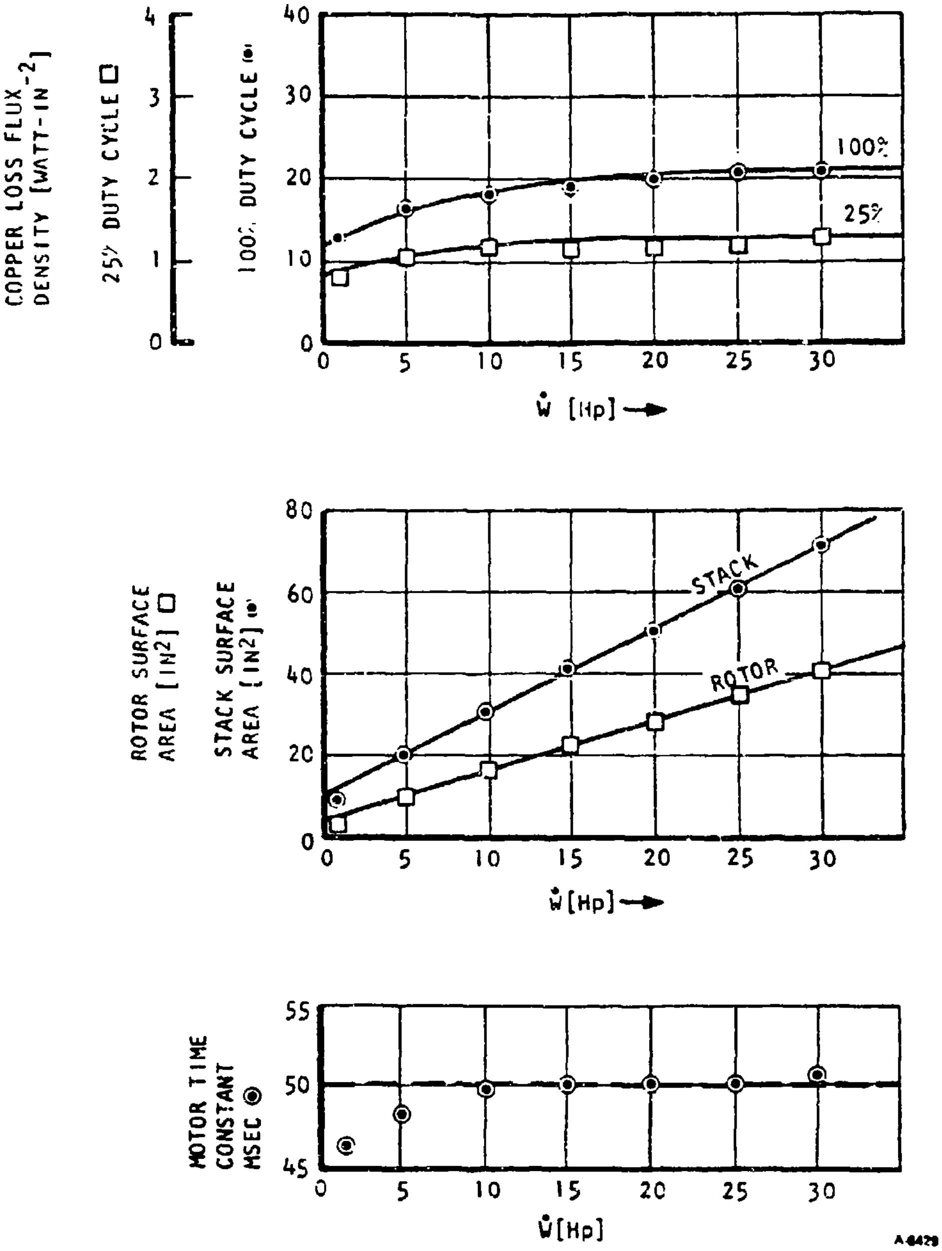


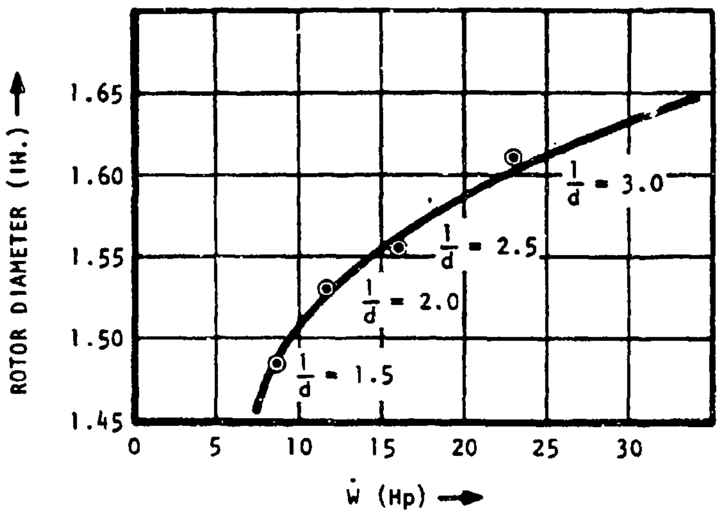
Figure D-8. Motor Data Summary

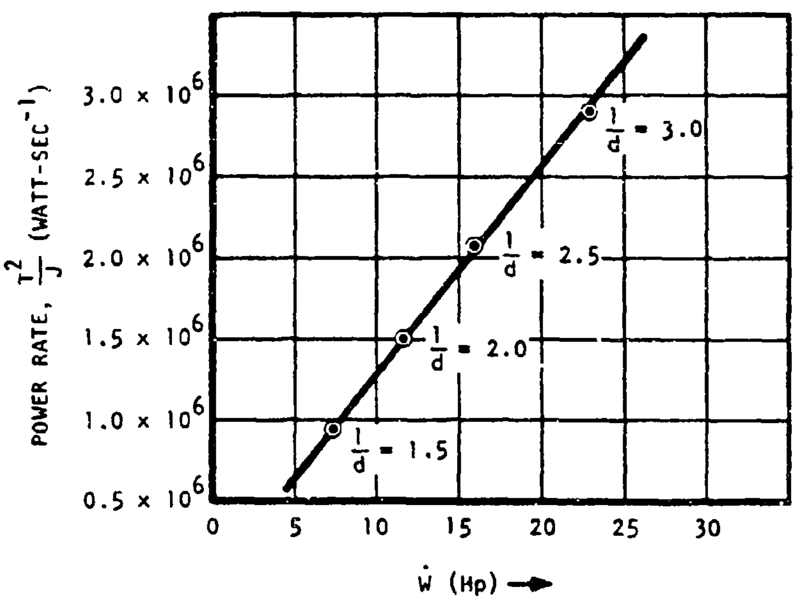
TABLE D-5

MOTOR DATA AS A FUNCTION OF l/d

Parameter	Ratio*, l/d (-) 1.5 (1.523)	2.0 (2.033)	2.5 (2.528)	3.0 (3.147)
Rotor Diameter (in.)	1.485	1.530	1.557	1.610
Housing Diameter (in.)	2.7	2.8	2.8	2.9
Housing Length (in.)	6.4	7.3	8.1	9.3
Motor Weight (lb)	3.0	3.7	4.8	6.5
Current Limit (amp)	26.6	41.4	56.3	80.6
Thermal Flux Density (watt-in.^{2})**				
• 25 percent	1.11	1.15	1.20	1.24
• 100 percent	17.7	18.4	19.2	19.8
Speed (krpm)***				
• A	23.3	23.3	23.3	23.2
• B, D	20.0	20.0	20.0	20.0
Torque (in.-lb)***				
• D, E	23.8	37.1	50.5	72.3
Time Constant (msec)*	50 (52.0)	50 (51.3)	50 (50.8)	50 (51.9)
Rated Power (hp)	7.55	11.76	16.01	22.96

*Specified (actual)
**Copper loss only
***See Figure A-4.





CONSTANT VARIABLES:

$\dot{\theta}_M$ = 20k rpm (POINT D)

τ_M = 50 msec (POINT E TO D)

A-6087

Figure D-9. Motor Data, Constant $\dot{\theta}_m$ and τ_m

TABLE D-6

MOTOR DATA AS A FUNCTION OF τ_m

Parameter	Time Constant (msec)*		
	50 (50.4)	75 (74.9)	100 (97.8)
Ratio*, l/d (-)	3.0 (3.089)	3.0 (2.969)	3.0 (2.695)
Rotor Diameter (in.)	2.949	2.385	2.239
Housing Diameter (in.)	4.8	4.0	3.7
Housing Length (in.)	14.0	11.7	10.5
Motor Weight (lb)	34.8	18.7	13.7
Current Limit (amp)	210.5	210.5	208.2
Thermal Flux Density (watt-in.^{2})**			
• 25 percent	1.35	1.25	1.41
• 100 percent	21.6	20.0	22.5
Speed (krpm)***			
• A	8.1	17.4	24.2
• B, D	7.0	15.0	21.0
Torque (in.-lb)***			
• D, F	524.4	252.8	181.9
Rated Power*	60. (58.2)	60 (60.2)	60 (60.6)

*Specified (actual)
**Copper loss only
***See Figure A-4

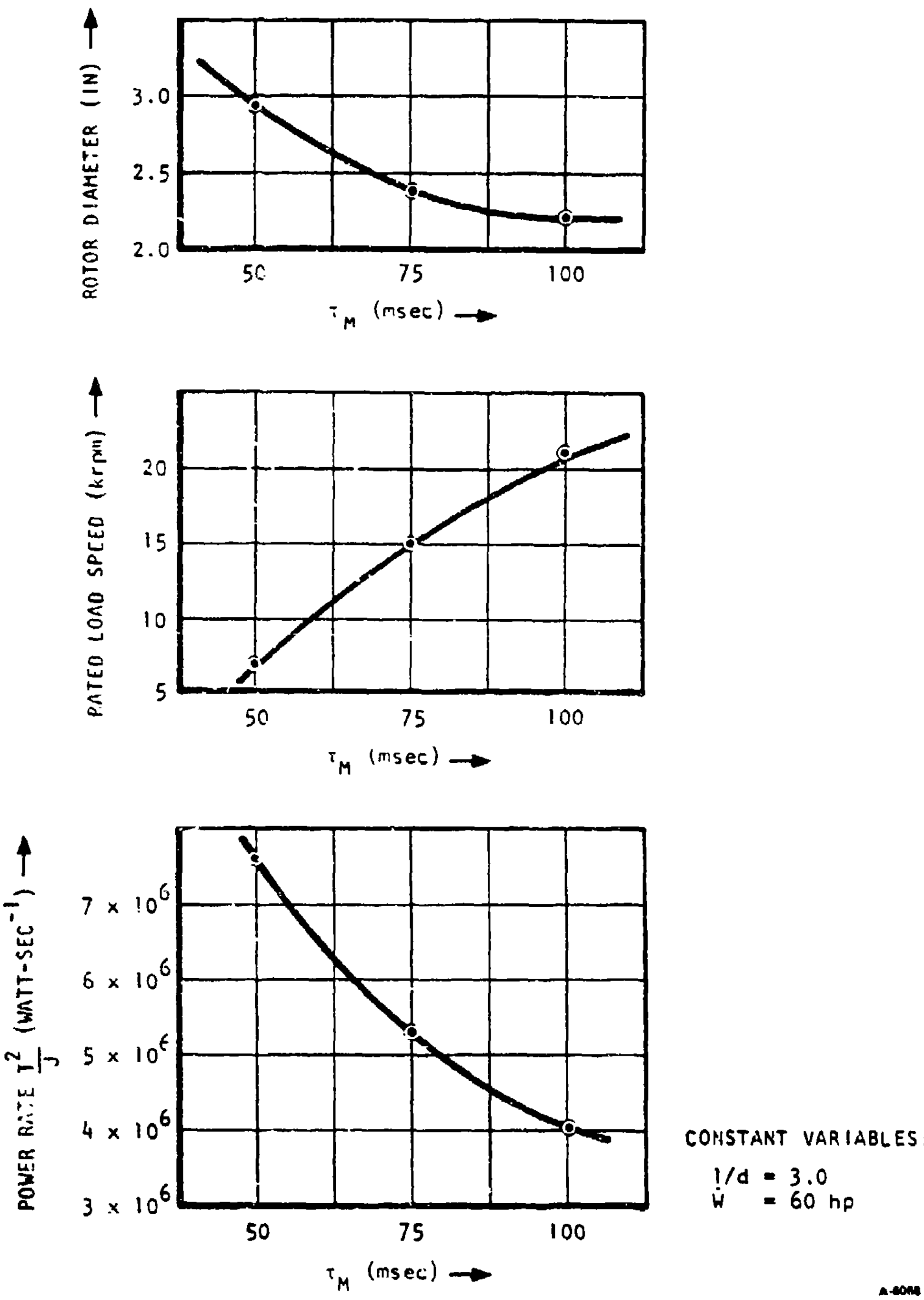


Figure D-10. Motor Data, Constant l/d, $\dot{W}$

A.3 THERMAL INVESTIGATION

A preliminary thermal analysis of one motor was performed as part of Task 3. The object of the analysis was to determine motor cooling techniques to assure that critical motor elements (i.e., permanent magnet, copper windings, etc.), remained within material temperature limits. A summary of the analytical results completed is presented below.

The motor selected for analysis was the 20 hp motor presented in Paragraph A-2. This motor was selected because its thermal flux density was representative of most of the motors generated. The motor was operated under the following duty cycle:

(a) A steady-state condition at 10% speed and 25% load.

(b) A two-minute transient condition immediately after case 1 at 100% speed and 50% load.

(c) A short time (less than 1 second) transient-state condition immediately after case 2 at 100% speed and 100% load.

The provisions above were used as a baseline condition for the thermal analysis, and were determined to be representative of the aircraft requirements.

There were two motor cooling techniques investigated in the analysis. The first implemented fins on the housing to increase the natural convection and radiation heat transfer to the surroundings (no forced air in the motor). The fins would be cast on the 1/8 inch thick aluminum stator housing, spaced 0.25-inch apart with a 0.04-inch fin thickness and a 0.5-inch fin height. Figure D-11 shows this finned housing configuration. A schematic diagram of the second cooling technique appears in Figure D-12. Cooling air is forced through the rotor to keep the Sa-Co (samarium-cobalt) permanent magnets at an acceptable temperature level. Figure D-13 shows a drawing of the rotor cooling flow passages: (1) rotor-stator gap-flow, (2) 6-32/32-inch hole-flow, and (3) rotor web shaft flow. This cooling scheme did not use fins on the housing. The motor thermal nodal network is also shown in Figure D-13. The first of each set of numbers represents the air inlet half of the motor, while the numbers in parenthesis represent the air outlet half. The Sa-Co permanent magnets (nodes 1 & 6) are contained by a sleeve (nodes 43 and 44). The stator stack windings are represented by nodes 11-12 and 26-27 while the end-turn and end-section windings are nodes 13-14, 28-29 and 15, 30, respectively.

The motor nodal model was prepared by means of the AiResearch Stator Armature, and LIM Thermal Model Generation Computer Program H0061 (Appendix A-2). The program connects each of the motor nodes by a conduction and convection resistance array such that the motor thermal model may be combined with the rest of the system. The complete model is then analyzed by the AiResearch Thermal Analyzer Computer Program H0910 (Appendix A-2).The latter program is capable of simulating conduction, convection and radiation calculations and other heat transfer mechanisms.

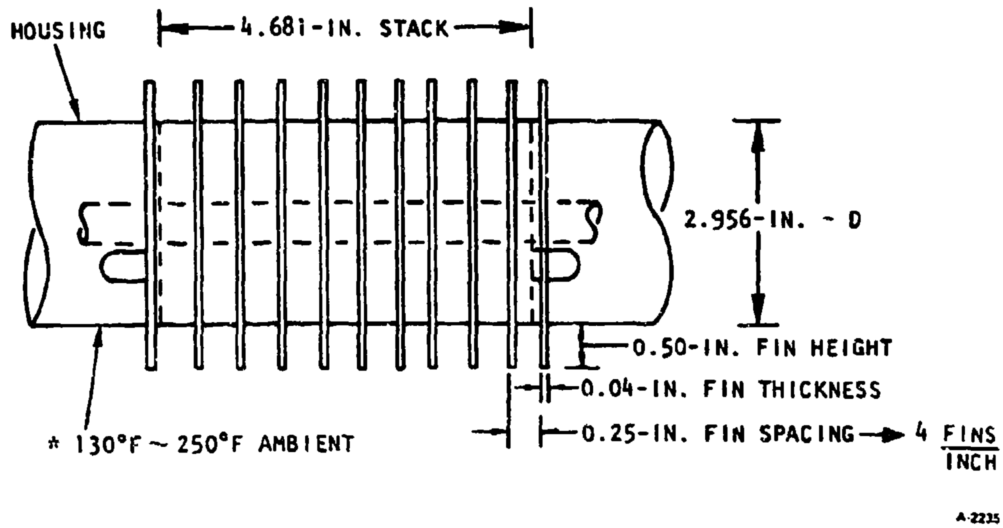


Figure D-11. External Cooling Scheme

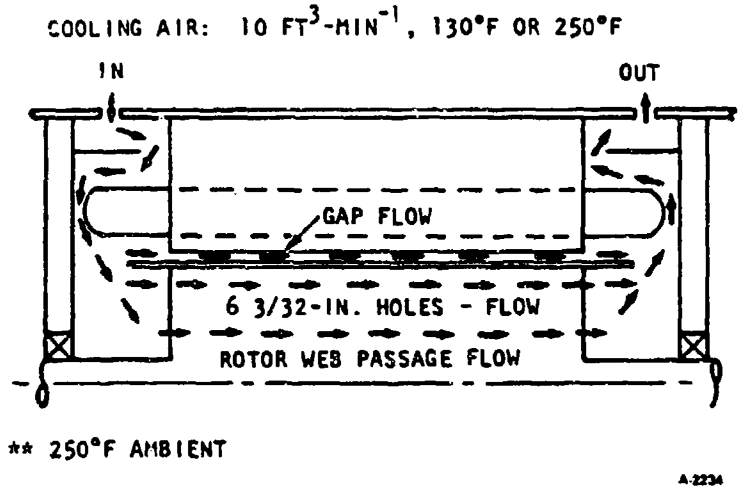


Figure D-12. Internal Cooling Scheme

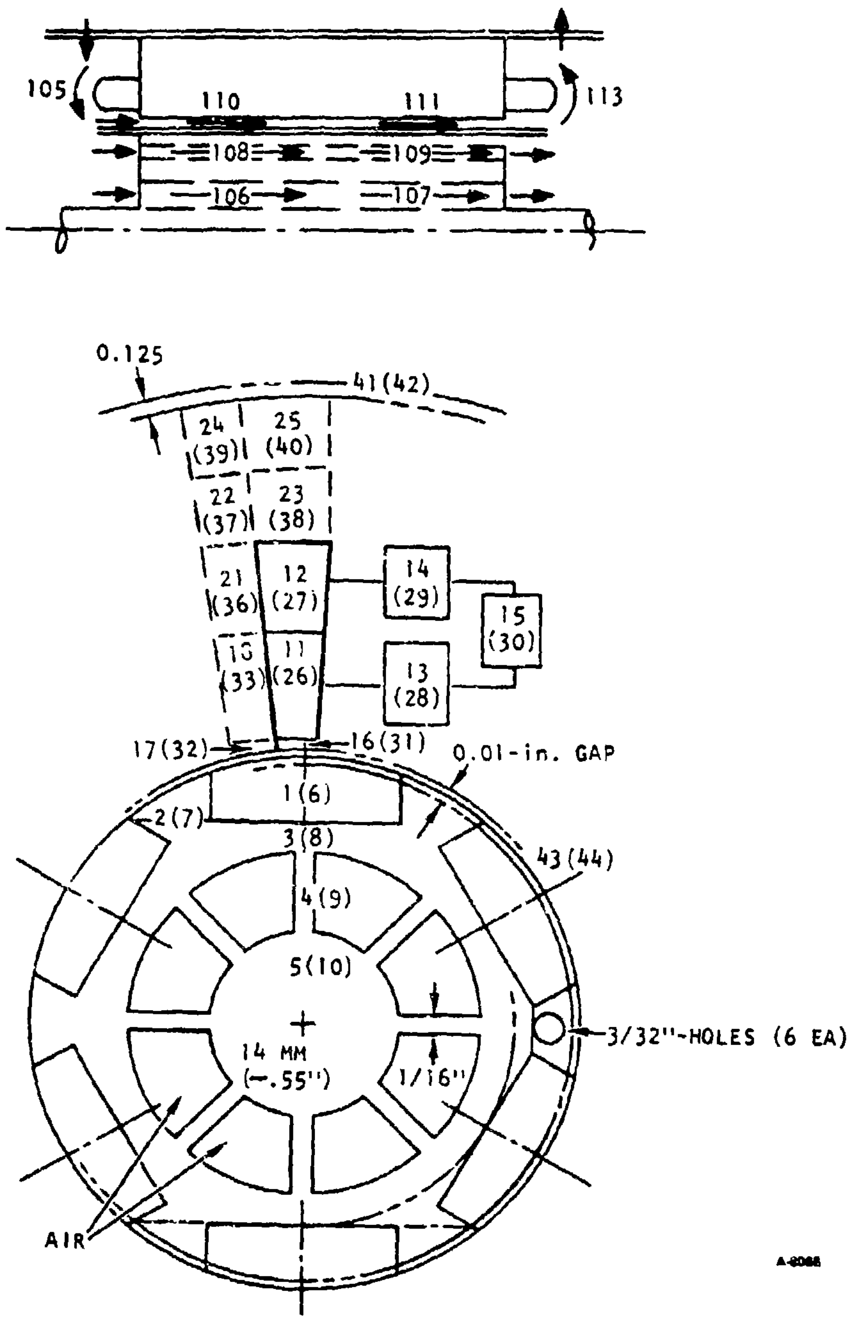


Figure D-13. Thermal Nodal Model

The losses of the 20 hp motor for the duty cycle are shown in Table D-7, and were obtained from the motor design program [12]. All entries in the table are in watts and copper losses are based on 300°F temperature. The actual copper losses used in the analysis, however, varied as function temperature depending upon the resistance characteristics of copper.

TABLE D-7

MOTOR LOSS SUMMARY

Condition	Motor Speed (rpm)	Copper	Tooth	Back Iron	Pole Head	Windage	Total
10% speed 25% load	1800	63.72	8.92	8.16	3.51	0.02	84.33
100% speed 50% load	18000	252.40	135.50	120.0	44.66	5.26	557.82
100% speed 100% load	18000	1021.30	127.80	113.20	115.56	5.26	1383.12

Notes: All losses are in watts
Copper losses are given at 300°F

The results of the thermal analysis for critical motor elements are tabulated in Table D-8. Entries under cooling scheme A and B utilized the finned housing configuration where motor cooling was achieved by natural convection and radiation to a 130°F and 250°F ambient, respectively. Cooling schemes C and D, on the other hand used forced air through the rotor to cool the motor with an air inlet of 130°F and 250°F, respectively, both with 250°F ambient. As noted in the table, temperatures under A1, B1, C1, and D1 depict steady state temperature predictions with 10 percent speed and 25 percent duty load under the respective cooling scheme. Entries under A2, B2, C2, and D2 are temperatures at the end of the two-minute transient state condition with 100 percent speed and 50 percent load, immediately after the steady state run. Likewise, temperatures under A3, B3, C3 and D3 are the transient response at indicated elapsed time with 100 percent speed and 100 percent load, immediately after the two-minute transient run.

Although the ambient temperature was specified to be close to 250°F, cooling scheme A with 130°F ambient was also analyzed to show the cooling effectiveness of the finned housing configuration. The time-temperature response of a complete cycle under cooling schemes B and C is shown in Figure D-14. The 250°F ambient and 130°F inlet temperature for forced cooling were used as a baseline condition that was specified. The volumetric flow rate in cooling schemes C and D was ~10 cfm with a 130°F inlet temperature. This corresponds to a maximum pressure differential of 0.70-in H_2O from the inlet to the outlet of the air stream shown in Figure D-12.

TABLE D-8

MOTOR ELEMENT TEMPERATURE SUMMARY

Element	Cooling Scheme - A*			Cooling Scheme - B			Cooling Scheme - C			Cooling Scheme - D		
	A1+	A2	A3(10 sec)	B1	B2	B3(2 sec)	C1	C2	C3(10 sec)	D1	D2	D3(5 sec)
Permanent Magnets (samariam cobalt)	226	257	264	345	378	380	205	249	257	308	353	357
Copper Windings	230	306	385	352	437	458	230	308	382	324	411	459
Top Tooth	223	276	295	343	401	402	226	298	318	321	396	405
Back Iron	219	262	270	338	385	386	229	298	309	322	394	399

*Cooling Schemes:

A - Finned housing, natural convection and radiation cooling to a 130°F - ambient
B - Finned housing, natural convection and radiation cooling to a 250°F ambient
C - Forced cooling air through the rotor with a 130°F air inlet temperature, no fins, 250°F - ambient
D - Forced cooling air through the rotor with a 250°F air inlet temperature, no fins, 250°F - ambient

+Conditions:

A1, B1, C1, D1, - Steady-state temperature predictions @ 10% speed - 1800 rpm and 25% load
A2, B2, C2, D2, - Two-minute transient temperature response @ 100% speed - 18000 rpm and 50% load
A3, B3, C3, D3, - Transient temperature response at indicated elapsed time @ 100% speed and 100% load

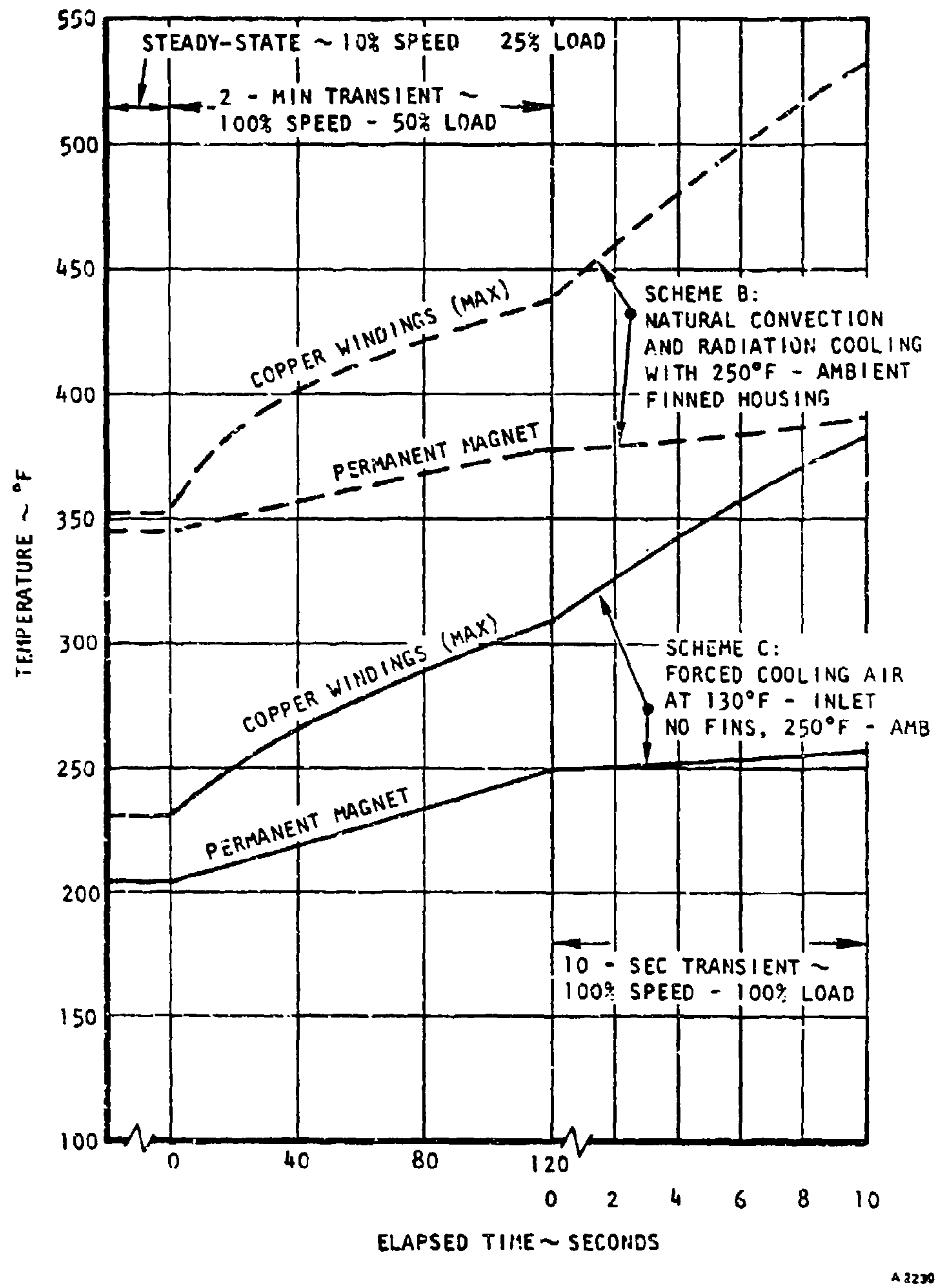


Figure D-14. Motor Temperature Profiles

A.4 CONCLUSIONS

Conclusions concerning motor design, performance, and selection were drawn during Task 3. These conclusions are presented in the following paragraphs. The reader should note that these conclusions are valid for the design criteria used.

A.4.1 Motor Time Constant

A given motor time constant limits allowable motor rated power or speed.

First, Figure D-9 indicates that if l/d is a design constraint, then there is a maximum power which can be achieved for a given rated speed and time constant. As an example, Figure A-8 indicates that for

$$\dot{\theta}_m = 20 \text{ Krpm}$$

$$l/d = 3$$

$$\tau_m = 50 \text{ msec}$$

then

$$\dot{W}_{max} \cong 30 \text{ hp}$$

Second, Figure D-10 illustrates that as time constant decreases, the motor rated load speed must decrease. For example (Figure D-10), if

$$\dot{W} = 60 \text{ hp}$$

$$\tau_m = 50 \text{ msec}$$

$$l/d = 3$$

then

$$\dot{\theta}_{max} \cong 7 \text{ Krpm}$$

. reason for these effects is that motor inertia and torque determine eration; and motor rated speed effectively determines τ_m for some celeration. Although many individuals associate high motor speeds response, the above indicate that the opposite may generally hold true. example, given two motors, the motor with the greater acceleration will not necessarily have the fastest response when coupled to a load through a reduction drive which provides a fixed no-load speed.

A.4.2 Motor Power Rate

Motor power rate is a near linear function of rated power or time constant.

Figure D-9 shows power rate as a function of rated power. For the design criteria used, power rate is an exceptionally linear function of rated power.

Note that rotor length-to-diameter ratio varies with motor power also; but this variation may be interpreted as the necessary rotor geometry to satisfy the fixed constraints. In this case, power rate may be construed as a direct, linear function of rated power.

By referring to Figure D-10, one sees that power rate is a nearly linear function of time constant; actually, for the presented data, a possible empirical form for power rate may be

$$\ddot{W}(\tau_m) = a\,\tau_m^{-n} \qquad \text{(A-14)}$$

$$n \geq 0$$

where

$\ddot{W}$ = motor power rate

τ_m = motor time constant

a, n = empirical coefficients

However, approximating power rate as a linear function of time constant over a limited range appears to be reasonable. This conclusion holds for the design criteria of the figure: constant rotor length-to-diameter ratio and power.

Power rate is sometimes used as a parameter to select motors for servo applications [1, 2, 7, 8, 9]. It is a convenient parameter for comparing various motors, and indicates the ability of a motor to accelerate a load as well as itself. Motor acceleration indicates only the ability of a motor to accelerate itself, and is useful primarily for no-load dynamic performance analysis.

A cursory derivation of motor power rate may be useful to the reader*:

$$\dot{W} = T_m\,\dot{\theta}_m \qquad \text{(A-15)}$$

where

$\dot{W}$ = motor rated power

T_m = motor rated torque

$\dot{\theta}_m$ = motor rated speed

now

$$\ddot{W} = \frac{d}{dt}\,\dot{W} = T_m\,\ddot{\theta}_m + T_m\,\dot{\theta}_m \qquad \text{(A-16)}$$

*Other deviations of power rate are presented in the literature [7, 9]

For the case of constant torque applicable for our definition of τ_m

$$\dot{T}_m = 0$$

so

$$\ddot{W} \triangleq T_m \ddot{\theta}_m = T_m \frac{T_m}{J_m} = \frac{T_m^2}{J_m} \qquad \text{(A-17)}$$

The above is defined as motor power rate.

A.4.3 Motor Weight

The importance of properly selecting a motor for minimum weight is emphasized by the data of Paragraph A.2.

Figure D-8 shows that motor weight increases in direct proportion to motor power. By referring to Table D-4 one sees, however, that nearly all of the motors are in the same rated speed range (the 25 and 30 hp motors are exceptions).

Examining Table D-6 provides a totally different trend: three different weights for the same rated power and different time constants. For this case, motor weight increases as rated speed declines and as motor time constant declines. This is a penalty associated with increasing dynamic response.

This comparison emphasizes the need to utilize a motor with a time constant (power rate) no smaller (larger) than necessary to satisfy actuator dynamic performance requirements.

A.4.4 Motor Thermal Response

All cooling schemes investigated were acceptable for the duty cycle and assumptions utilized. Thus, no motor cooling would be required for any of the FCS actuation systems.

All temperature entries tabulated in Table D-8 are acceptable temperature levels for the different critical motor elements. To indicate the magnitude of temperature response during the 100-percent speed and 100-percent load condition, the elapsed times noted in the table are longer than the required operation time of less than 1 second. Temperature reponse at these conditions and at this short time interval run very close to that at the end of the two-minute transient condition with 100-percent speed and 50-percent duty load.

It should be noted that if higher conductor current densities and current limits had been used, it is likely that motor cooling (such as cooling scheme C) would have been required for some of the FCS applications. Thus, the assumptions developed in Paragraph A.1.4 have greatly simplified the actuation system design process.

APPENDIX A-2

COMPUTER PROGRAMS

COMPUTER PROGRAM BIGMAG

The Garrett Corporation has developed a computer program to facilitate the design and evaluation of certain types of permanent magnet machines interfacing with converters or with conventional linear systems. This program is called BIGMAG. Generally, only a few basic inputs have to be specified by the user. BIGMAG then synthesizes a baseline design suitable for optimization studies.

ELECTROMAGNETICS AND LOSSES

Classic salient-pole, two-reaction, analysis of synchronous machines has been adapted to meet the special cases of tangentially magnetized and radially magenetized permanent-magnet (PM) rotors. The magnetic circuit is represented by an equivalent circuit in which the iron and leakage paths and the magnet and stator MMF's are represented by lumped parameters. Nonlinearity of the iron reluctances vs flux density is taken into account by a look-up routine using B-H data tables for a variety of magnetic and PM materials. Special routines account for configuration effects (i.e., tapered teeth, shaped magnets). The flux leakage paths are estimated by special methods evolved from computerized flux plots, by classic field analysis of elemental situations, or by other methods found in the literature.

Losses calculated include: iron losses, pole head and damper head losses, stray load loss, copper and hysteresis losses, and windage losses.

STRUCTURAL ANALYSIS

The structural analysis predicts damper hoop stress, average compressive stress of the magnet, pole root stress, and stress in the non-magnetic hub which supports the poles and magnets. All geometric details are taken into account and a table of stress concentration factors and a look-up routine are included.

THERMAL ANALYSIS

Thermal analysis is limited to determining a credible stator copper current density to attain a specified final copper termperature with a given fluid, fluid pressure drop, and duty cycle. (Conversely, temperature will be determined if current density is specified.) In addition, certain elemental thermal data are given such as watts/lb in tooth and core iron; watts/sq. in. pole head and damper losses; and watts/sq. in. stator iron loss distributed over the outer periphery of the stack.

WEIGHT ANALYSIS

The weight of the basic electromagnetic parts is calculated. In calculating rotor weight, vent holes in the poles and under the magnets are taken into account. Also, the weight of the nonmagnetic hub supporting the poles

is included. To account for shaft extensions, bearings, end frames, stator frame, terminals, and fittings, a weight target equal to 20 percent of the electromagnetic weight is used to estimate total weight.

RECTIFIER/CONVERTER/INVERTER ANALYSIS

In addition to linear three phase systems (such as a linear load impedance or power supply connected to the machine), thyristor converters and transistor inverters may be modeled.

STATOR, ARMATURE, AND LIM THERMAL MODEL GENERATION COMPUTER PROGRAM (H0061)

A computer program has been developed which prepares a thermal model of an ac motor or generator stator, a dc motor armature, an ac generator field winding, or a linear induction motor. The program constructs a thermal model in the format of the AiResearch H0910 or H0298 thermal analyzer computer programs. This thermal model may then be combined with the thermal model of the rest of the system to form a complete motor, alternator, generator, turbo-alternator, or flywheel/motor/alternator.

The input and options to the geometry program are available in the output of the AiResearch motor, generator, and linear motor design and performance programs. Where possible the nomenclature and variable names have been preserved to provide easy and accurate model construction. The program has options provided to construct the thermal model of the following:

- One stack and end turn element model
- One half stack and end turn element model
- Two half stack and end turn element models connected in the center

Options are also provided to consider a constant width slot with rectangular windings or a constant width tooth with round wire windings. Cooling holes may be considered in the back iron arranged in either a square or triangular spaced pattern. The end turns may be considered with separate insulation surface elements for forced convection cooling through the end turns or with only the copper windings elements including conduction heat transfer between the windings and from the end turns into the stack.

THERMAL MODEL

The thermal model of the stack is constructed with winding elements in the upper and lower half of the slots. Each stack winding is connected to corresponding end turn elements and corresponding elements in the tooth by conduction. The stack winding elements are also connected to adjacent stack winding elements, the top stick element, the center stick element, and the back iron element at the base of the slot. The end turn winding elements are connected to corresponding end section elements and to adjacent end turn winding elements. Separate insulation surface elements may be generated for each end turn and end section winding element which are connected to a fluid stream element by forced convection and to the winding element by conduction. The end section winding elements are connected to both the upper and lower half end turn elements by conduction. The tooth elements are connected to adjacent tooth elements with the top of the tooth connected to the top stick, the center of the tooth connected to the center stick and the bottom of the tooth connected to the back iron element at the base of the tooth by conduction. The top of the tooth and the top stick are connected to the air gap by forced convection of a rotating cylinder in a static housing.

The back iron is divided into NBAK pairs of elements plus NROW elements with cooling holes punched in them. Each of the NBAK pairs of elements includes elements under the slot and under the tooth. These elements are connected to each other by conduction and the NROW elements are connected by forced convection to fluid stream elements in the holes. The other elements of the back iron are connected to element NOUT by conduction.

The initial temperature, heat dissipation, density, volume, specific heat, and thermal conductivity are computed and assigned to each element in the model. The total winding copper losses are divided in proportion to length and number of conductors between the stack, end turn, and end section winding elements. The surface losses are applied to the top of the tooth and the tooth losses are distributed equally to each tooth element adjacent to a stack winding element. The back iron losses are proportioned according to volume in the NBAK elements and the remaining back iron losses are distributed equally over the NROW elements.

As the thermal model is generated the program checks for consistency of the input data by calculating the area available in the slot for the copper conductors and comparing it with the input value of conductor cross-sectional area. When these areas match, the thermal model generated can be punched out by setting TAPE = 1.0.

H0910

STEADY-STATE AND TRANSIENT THERMAL ANALYZER PROGRAM WITH COMPRESSIBLE AND INCOMPRESSIBLE PRESSURE DROP AND FILM AND TRANSPIRATION COOLING

PROGRAM CAPABILITIES

Transient Heat Transfer Calculations

Transient heat transfer calcualtions are developed by an explicit finite difference technique using any element shape with three-dimensional conduction, convection, or radiation heat transfer.

Steady State Heat Transfer Calculations

Steady-state heat transfer calculations are based on a modified Gauss-Seidel solution to the simultaneous equations in the thermal model. This modified technique involves "accelerated" step substitution with monotonic deceleration until successive substitutions are convergent. A method of "lumping" areas of the problem which are slow to converge is also used to accelerate the calculation procedure. This procedure also provides for any element shape with three-dimensional conduction, convection, or radiation heat transfer.

Conduction Heat Transfer Calculations

Conduction heat transfer is input to the program by specifying the element numbers connected by conduction, the cross-sectional area for conduction between the elements, and the conduction length from the center of each element to the interface between them. A mechanical joint thermal contact resistance may also be specified between the elements if they are mechanically separated at the interface. The program obtains the thermal conductivity of each element from a table in which it may be specified as a constant value or as a function of temperature.

Convection Heat Transfer Calculations

Convection heat transfer is input to the program by specifying a solid element number connected to a fluid element number by convection, the cross sectional area for convection from the solid element, and the conduction length from the center of the solid element to the convection surface. This program performs the important and often overlooked task of combining conduction heat transfer from the center of the solid element to the surface with convection from the solid surface to the fluid.

The convection heat transfer coefficient may be input to the program by nine different methods. In the first four methods, the heat transfer coefficient may be input as a constant, as a function of time in a table, as a

function of the surface to fluid temperature difference in a table, and as a function of the "film" temperature in a table. In method five, the program calculates the natural convection heat transfer coefficient for both open and enclosed static spaces and enclosed rotating spaces. In method six the program calculates convection heat transfer coefficients for high speed laminar or turbulent flow over external surface including the effects of the "recovery" temperature in the boundary layer. In method seven the program calculates convection heat transfer coefficients on a free or enclosed rotating disc including the calculation of frictional "windage" heat generation. In method eight the program calculates jet impingement heat transfer coefficients for impingement from a row of holes onto a concave surface. In method nine the program calcualtes convection heat transfer coefficients for flow in a duct, including the heat transfer "fin effectiveness" of extended surfaces within the duct. This method utilizes tables of Colburn J-factors input as a function of Reynolds number to the program. These tables may be generated for fluid flow in round ducts, square ducts, rectangular ducts, triangual ructs, annular spaces, dimpled tubes, and curved ducts. They may also be generated for fluid flow in tube banks, plate-fin surfaces, pin-fin surfaces, screen matrix surfaces, crossed rod matrix surfaces, and corrugated ceramic surfaces. Entrance effects on heat transfer may be applied using the appropriate multiplying factor at each location. Four techniques for evaluation of the influence of temperature-dependent fluid properties are available in the program. The appropriate fluid peoperties may be input in tabular form as a function of temperature.

Radiation Heat Transfer Calculations

Radiation heat transfer is input to the program by specifying a solid element number connected to a representative surrounding element number by radiation, the cross sectional area for radiation from the solid element, and the conduction length from the center of the solid element to the radiation surface. This section also includes the important combination of conduction to the radiating surface with radiation from the surface. The emissivity view factor for radiation may be estimated by methods given in "Radiation Heat Transfer" by Sparrow and Cess or by a computer program such as CONFAC II.

Initial Temperature, Boundary Conditions, Heat Input, Thermal Capacitance, and Thermal Conductivity Specification

The initial temperature, boundary conditions, heat input thermal capacitance, and thermal conductivity may be specified for each individual element or for blocks of elements which are identical. In transient heat transfer calculation, the initial temperature, the heat input, the density, the volume, the specific heat, and the thermal conductivity of each element is specified. For elements with negligible thermal capacitance the density, volume, and specific heat may be left blank to increase the calculation time step. For steady state calculations, the initial temperature, the heat input, and the thermal conductivity of each element is specified. The boundary condition elements are specified by having a negative value for the density times the volume. This element is then maintained at a constant temperature or may be specified as a temperature versus time function from an input table. Any element in the network may be specified as a boundary condition (constant

temperature) element and any number of elements may be connected to it by conduction, convection, or radiation. The heat input for each element may be input as zero, as a constant value, as a function of time in a table, as a function of its own temperature or another specified element temperature, specified from the frictional "windage" heat generation calculations, or calculated from the ball and roller bearing heat generation calculation computer program which can be supplied. The specific heat and thermal conductivity of each element may be specified as a constant or as a function of temperature in tables.

Fluid Stream Heat Transfer and Pressure Drop Calculations

Fluid stream elements may be input with heat transfer to them by conduction, convection, or radiation. Fluid stream heat transfer calculations have provisions for preventing the outlet fluid temperature from "overshooting" the surrounding surface temperatures, a thermodynamic impossibility. The steady state fluid stream calculations are based on thermal capacity rate calculations, while transient fluid stream calculations may be based on the thermal capacitance of each element moving in the fluid stream to simulate "lag" conditions. The energy input of rotational flow may also be added to the fluid stream.

Both steady state compressible and incompressible fluid stream pressure drops may be calculated by the program. The pressure drop calculations include the effects of heat addition, area change, fluid friction, rotational flow, and flow addition or removal. Total head losses due to valves, bends, sharp contractions or expansions, and orifices may be included at the inlet and exit to each fluid stream.

A complete fluid stream network may be simulated with streams branching from previous streams and mixing to form new streams or even returning to a previous stream in the network. The fluid flow rate may be input as a constant, as a function of time, from a table, or as a function of specified element temperature.

Film Cooling and Transpiration Cooling Calculations

Film cooling calculations have been included in a table of the film effectiveness as a function of the film cooling parameter (x/ms). Local film temperatures on a film cooled surface are calculated by the program from specified element temperatures for the "free stream" and for the film coolant discharge point. Local film temperatures are calculated at specified distances downstream from the point of film injection. The table of film effectivenss as a function of the film cooling parameter (x/ms), may be selected from twelve correlations presented in the transpiration and film cooling effectiveness computer program which can be supplied. The appropriate correlation should be selected for the method of film injection used.

Transpiration cooling calculations may be included by using the Stanton number reduction factor for transpiration cooling from the transpiration and film cooling effectiveness computer program or from papers on transpiration cooling. The thermal analyzer program accounts for the effect of cooling air discharge temperature from the transpiration cooled wall on the Stanton number reduction factor.

PROGRAM OUTPUT

1. Each element temperature, heat input, and thermal conductivity for steady state calculations is printed out. Each element temperature, heat input, weight, specific heat, and thermal conductivity for each specified printing time period in transient heat transfer calculations is printed out.

2. The fluid stream inlet temperature and the outlet temperature, the fluid stream flow rate, the fluid density, and the internal fluid heat generation for each section of each fluid stream is printed out.

3. The "free stream" temperature, the film discharge temperature, and the effective film temperature at each location specified is printed out.

4. The printing of the thermal resistance values for conduction, the thermal resistance values and heat transfer coefficients for convection, and the thermal resistance values and effective heat transfer coefficients for radiation may be included or deleted as specified.

5. The fluid stream pressure drop calculations and printout may be deleted if specified. When included, the total and static pressures, the Reynolds number, the friction factor, and the Mach number for compressible flow is printed for each element in each fluid stream.

TYPICAL APPLICATIONS

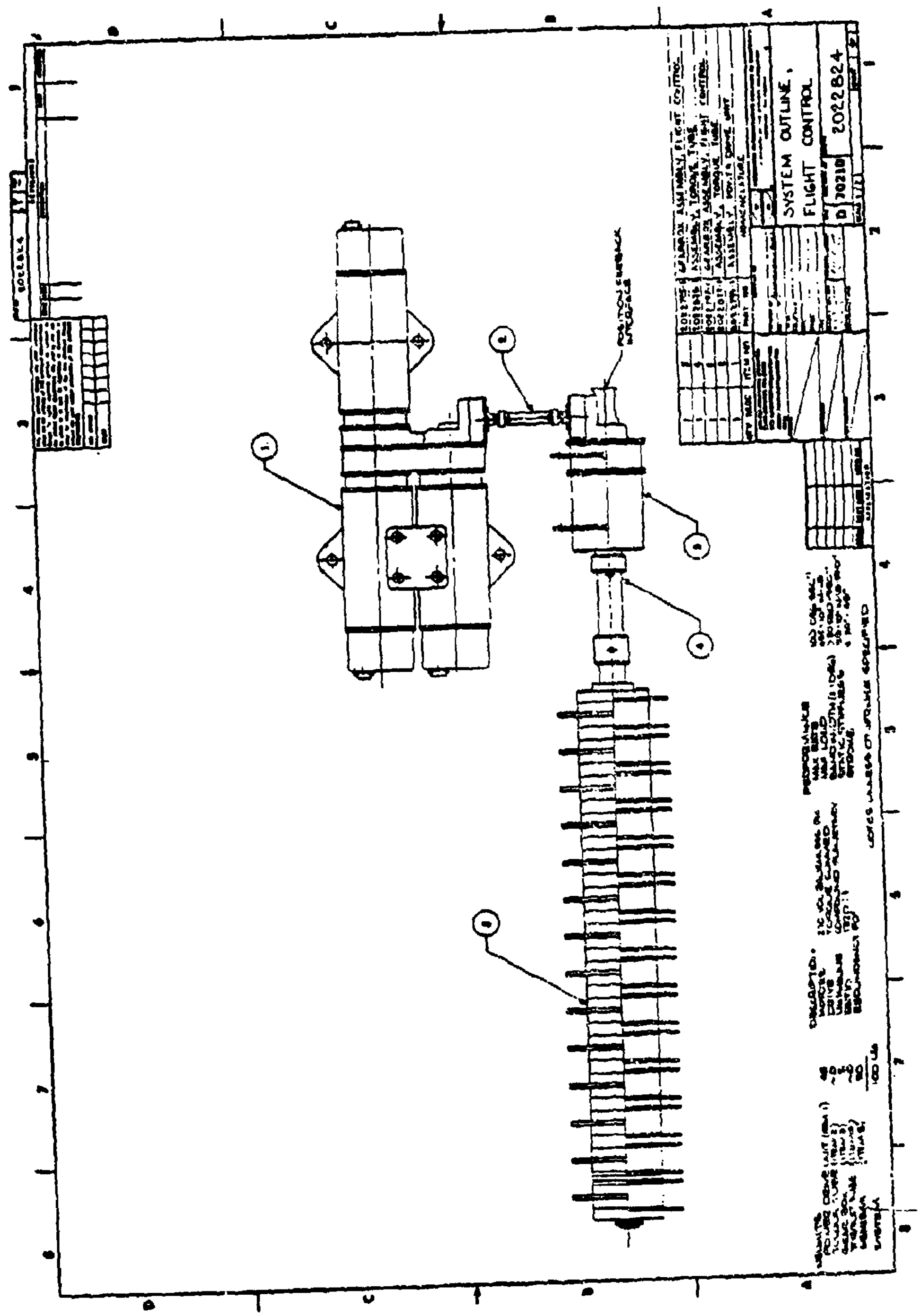
1. Both passive and active electronic cooling system analysis and design with or without heaters or cooling flow controllers.

2. Thermal analysis and design calculations for ambient cooled, forced air cooled, gas cooled, or liquid cooled ac or dc motors, generators, and alternators.

3. Thermal analysis and design calculations for pumps, fans, and compressors including the bearing temperatures and the analysis of the motors for turbines driving them.

4. Thermal analysis and design calculations of gas turbine engines including the axial flow and radial flow compressors and turbines, the combustor, the bearings and seals, the anti-icing system, the lubrication cooling system, the fuel supply system, and the accessory area cooling system. Also the thermal analysis of cooled and uncooled turbine blades.

5. Transient and steady state thermal analysis of heat exchangers including air-oil coolers, fuel-oil coolers, recuperators, rotary regenerators, cryogenic heat exchangers, pool boiling heat exchangers, condensers, periodic flow regenerators, and heat exchangers with more than two fluid streams. The calculations may include the effects of axial conduction, fluid bypassing, perfectly mixed or unmixed fluids, variation of fluid properties through the heat exchanger, condensation of moisture from the air or "wet" heat transfer, and the effect of the variation of fluid to wall temperature difference on local heat transfer coefficients for boiling and condensing.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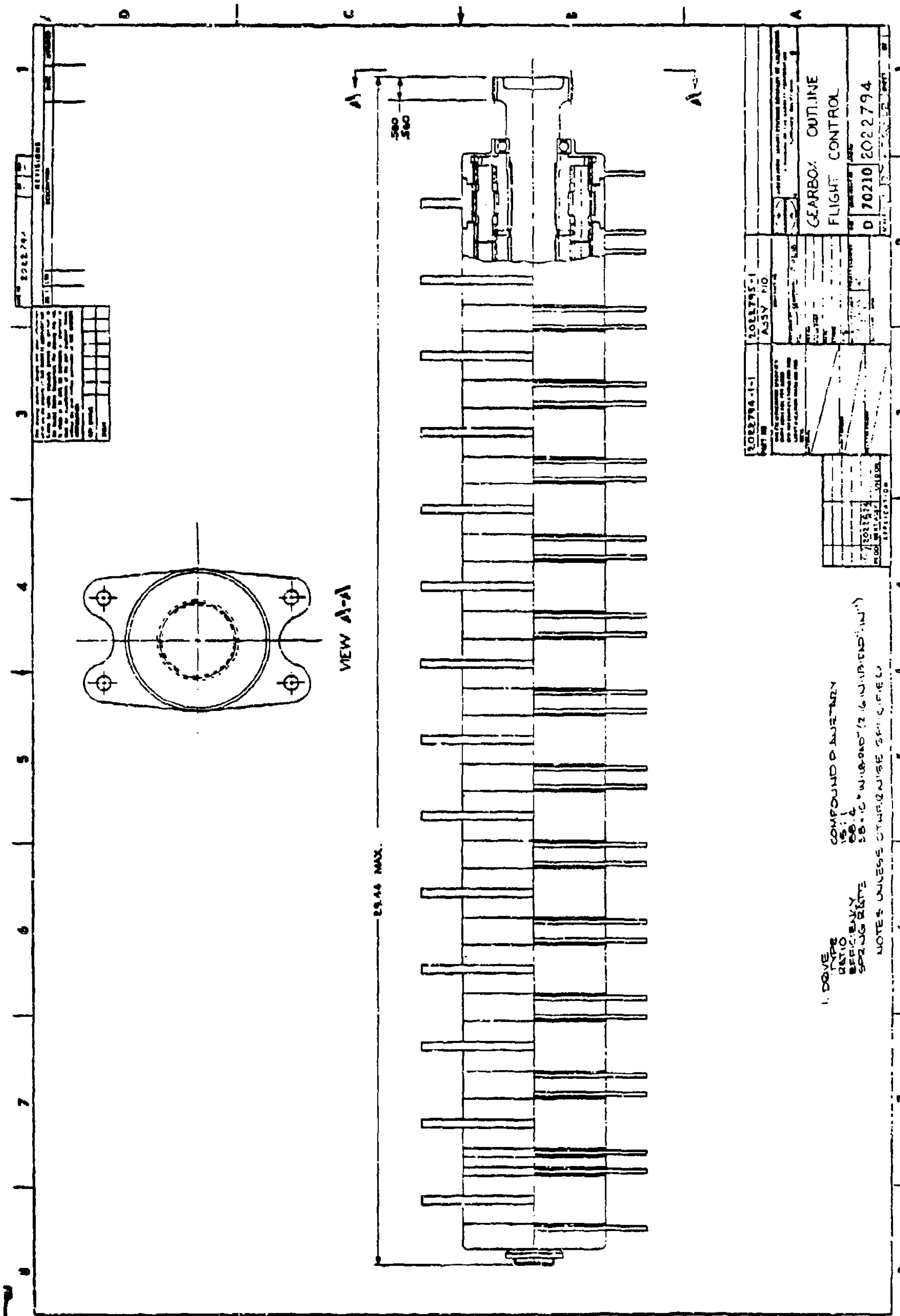
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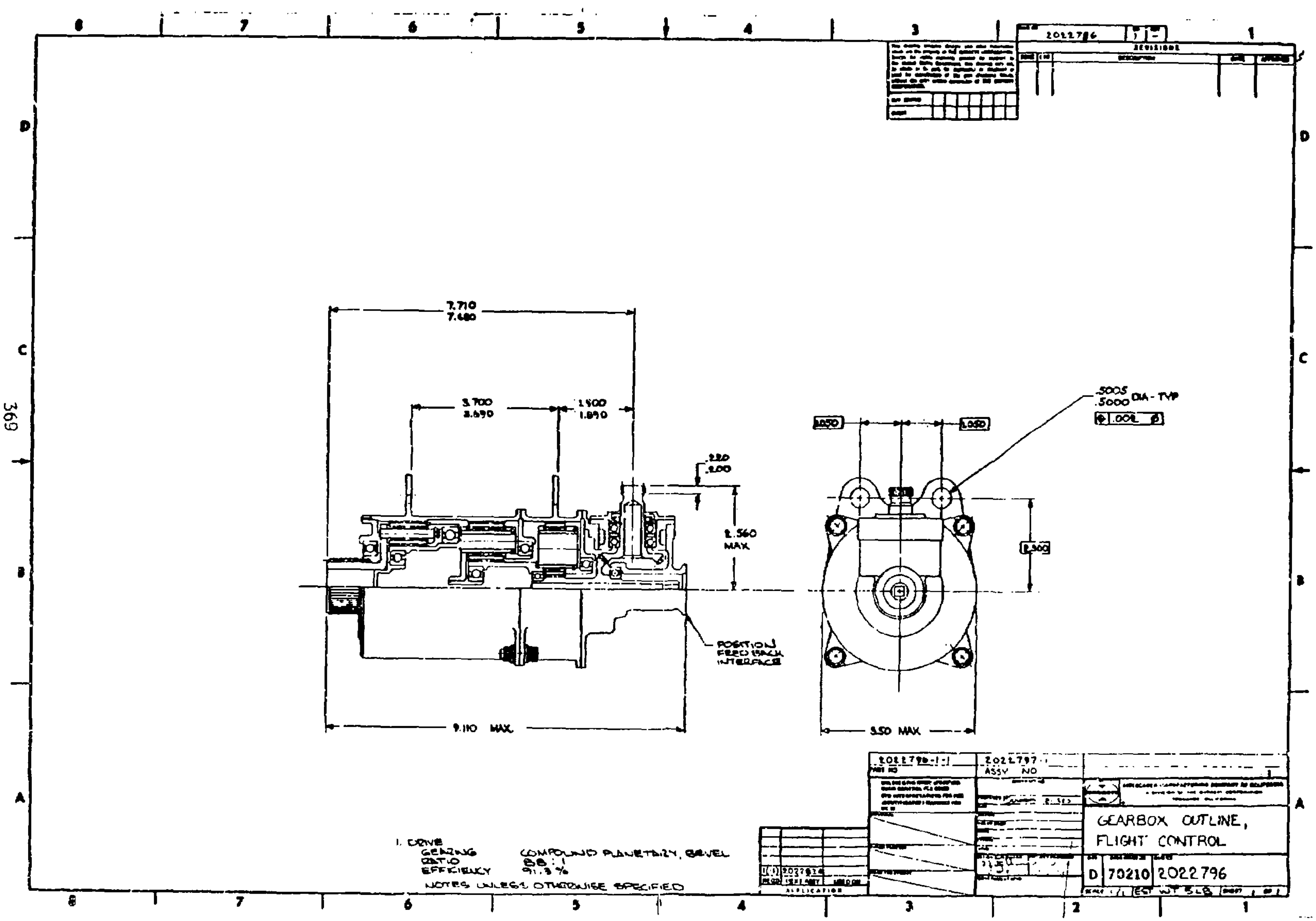
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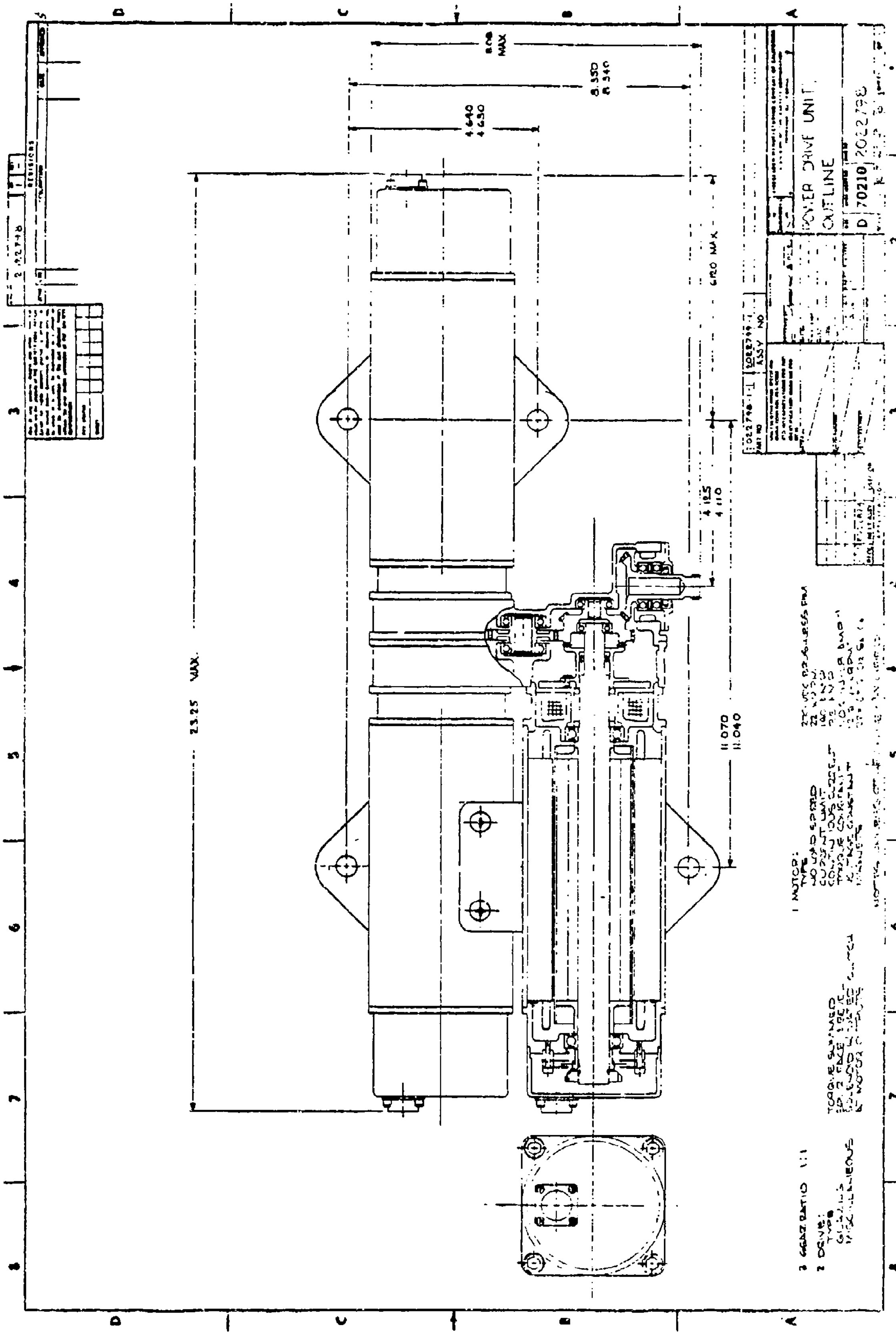

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